LISA-SC-DD-0001

Laser Interferometer Space Antenna (LISA)

Sciencecraft Description

Original

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NOTE:

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1 Introduction

1.1 PURPOSE

The purpose of the LISA Spacecraft Description Document is to provide engineers, scientists, and program management with the latest LISA Spacecraft design information available.

1.2 SCOPE

This document focuses on the NASA version of a LISA Spacecraft design that meets the mission architecture, science requirements and accommodates the ESA proposed “payload”. In addition to the science background and hardware element sections, a section on assembly, integration, and test (AIT) is included due to its integral importance to the Spacecraft design and planning.

2 Mission Summary

LISA is a joint European Space Agency (ESA)-NASA project to design, build and operate the first space-based, gravitational wave observatory. The design concept is based on monitoring changes in distance between proof masses in three spacecraft, orbiting the Sun in an equilateral triangle formation with 5 million kilometer arm lengths. The constellation of LISA "Sciencecraft" acts in concert, as "the instrument" to detect gravitational waves in the frequency band 3×10^{-4} to 10^{-1} Hz. Table 2-1 provides a summary of the baseline mission parameters. LISA directly probes the most extreme situations in the universe, most of which are difficult, or impossible, to observe with conventional electromagnetic observations.

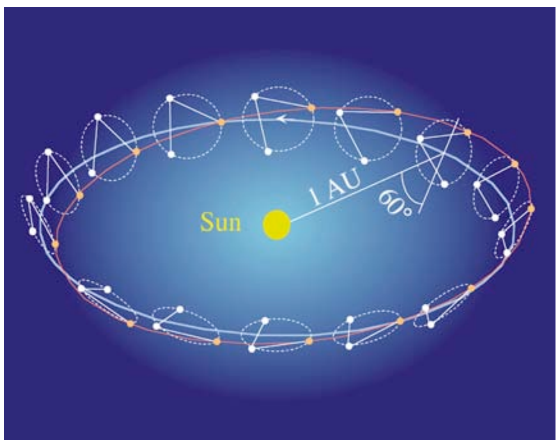
Table 2-1: Mission Baseline Parameters

Parameter	Comments
Lifetime	+ 5 year science operations phase (10 year goal) after 18 month cruise phase (14 months for transfer trajectory + 4 months commissioning)
Orbits	3 independent, Heliocentric, 20° earth trailing orbits, equilateral triangular constellation with 5 x 10^6 km +/- 1% arm lengths, constellation requires no active station keeping or maintenance over the mission lifetime
Launch Vehicle	Atlas V series, C3=0.5 km^2/s^2
Communications	Ka-Band – (2) HGA and (4) X-Band Omnis, 90 kbps downlink, 2 kbps Up DSN 34 m dish
C&DH	Supports Sciencecraft functions only, science data processing is performed on the ground
GN&C	Star trackers, sun sensors, gyros
EPS	Fixed SA, triple junction Gallium Arsenide (GsAs), 957 W EOL @30° Sun Angle, 766 W required w/30% margin; 9Ah, LiIon Battery, 60% DoD max.
Thermal	Micro-Kelvin stability with passive design
Mechanical	Bus is built around the Payload, Sciencecraft nests in Propulsion Module (P/M), 3 S/C are stacked in the fairing with the P/M carrying the majority of the launch loads
Propulsion Module	1100 m/s for primary burns, + 30 m/s for correction maneuvers delta V

2.1 MISSION CONCEPT

LISA measures time-varying strains in space-time by interferometrically monitoring changes in 5 million kilometer baselines. The three baselines extend between three spacecraft orbiting the Sun in a formation 20° behind the Earth as shown in Figure 2.1-1. The orbits are chosen to keep the three baselines as close to equal as possible over the mission lifetime. The Sciencecraft at the corners house two proof masses and interferometry equipment.

The three baselines form a nearly equilateral triangle that appears to cartwheel around the Sun once per year. The measured baselines extend from a proof mass in one spacecraft to another proof mass in a distant spacecraft. Hence the proof masses are the measurement fiducials defining the endpoints of the monitored distance. The orbits of the three Sciencecraft are identical except for the phasing of their inclinations. The plane of the triangle is inclined 60° to the Earth's ecliptic plane. This geometry has the added benefit of a very benign environment, and a constant solar illumination angle on the Sciencecraft, thereby reducing unwanted disturbances.



The LISA Sciencecraft orbits do not require any regular adjustments to maintain the formation throughout the life of the mission. The Sciencraft are represented by 3 dots in the snapshots of the formation's annual motion around the Sun. The orbit of one Sciencecraft is traced by the inclined circle running through the same dot in each snapshot.

Figure 2.1-1: LISA Sciencecraft Orbits

The proof masses are protected from disturbances by careful design and "drag-free" operation. In drag-free operation, the mass is free-falling, but sensors in the housing around the proof mass sense the relative position of proof mass and Sciencecraft, and a control system commands the Sciencecraft thrusters to follow the free-falling mass. This can be done with two proof masses, following each in only its sensitive direction. Drag-free operation keeps force gradients arising in the Sciencecraft from applying time-varying disturbances to the proof masses.

The distance measuring system is essentially a continuous interferometric laser ranging scheme. Lasers at each end of each arm operate in a "transponder" mode. A beam is sent out from one Sciencecraft to a distant one. The laser in the distant Sciencecraft is phase-locked to the incoming beam and returns a high power phase replica. When that beam returns to the original Sciencecraft, it is beat against the local laser. Variants of this basic scheme are repeated for all long baselines, and the lasers illuminating different baselines are also compared. Optical path difference changes, laser frequency noise, and clock noise are determined.

The disturbance spectrum and the noise floor of the ranging system conspire to give a useful measurement bandwidth from $3x10^{-5}$ to 1 Hz. The three arms can simultaneously measure both polarizations of quadrupolar waves. The source direction is decoded from amplitude, frequency, and phase modulation caused by annual orbital motion.

2.1.1 Science Orbit

Critical to the LISA mission is the selection of the science orbit as shown in Figure 2.1-2. A heliocentric orbit is chosen for the science phase of the mission primarily to minimize changes in the differences of the arm lengths over the operational life of the mission, without the need for orbital formation maintenance. The orbit selection also provides the benefits of: a thermally benign payload environment; minimized non-gravitational perturbations to allow for accurate micro Newton propulsion control; passively maintaining arm lengths; and a communications distance that allows adequate link margins using standard subsystem components.

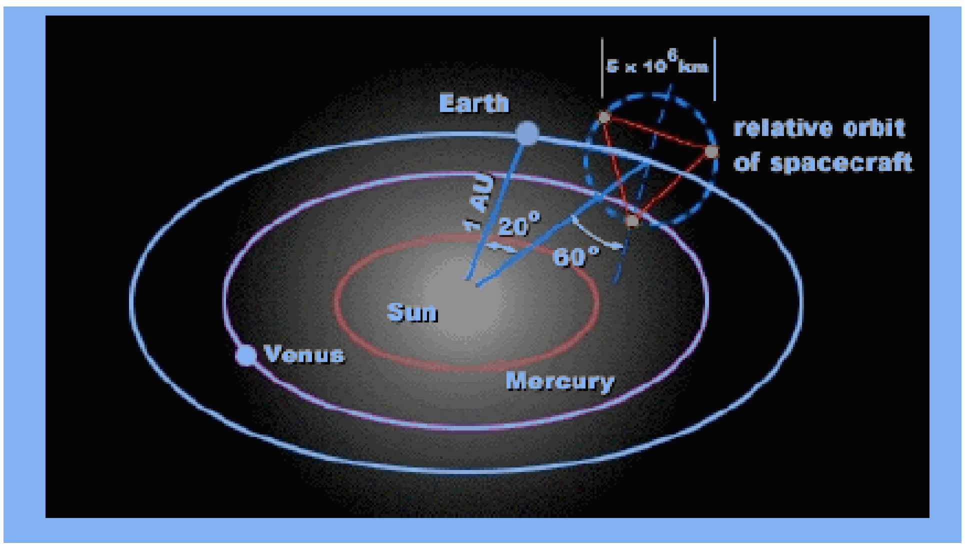


Figure 2.1-2: LISA Science Orbit

2.1.1.1 Orbit Constellation

The LISA constellation is based on an interesting, but little known situation in orbital mechanics. Start with a point in circular orbit about a body and make that point the center of a plane, which is tipped 60 deg (either up or down) from the central body and from the plane of the orbit. Keeping the plane centered on the orbiting point, revolve the plane with the point around the central body so it stays fixed in orientation relative to the central body. At the same time and at the same rate rotate the plane in that tipped orientation around the orbiting point so that the half nearer the central body is moving in the direction of the orbital motion. Thus in each orbit the plane revolves once around the central body and rotates once around the center point, where the axis of that rotation is the normal to the plane which is "down" relative to the plane of the orbit. Then as they follow their individual orbits around the central body, all the points on the plane are fixed points to first order in their distance from the center point (i.e., deviations from initial positions on the plane vary quadratically with the distance from the center point), ref. Figure 2.1-1.

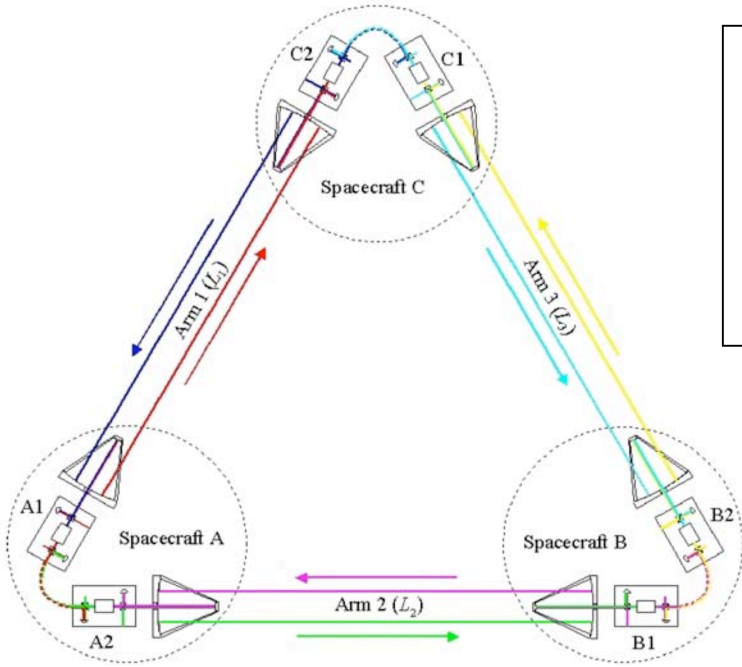
The LISA constellation is an equilateral triangle in this "fixed" plane. It is centered on the center point of the plane to minimize the second order variations in position and the differences in those variations. To stay approximately a fixed distance from Earth, the center point is in a 1 AU orbit; this center point orbits in the ecliptic plane to minimize the transfer ΔV; the center is about 22 deg behind the Earth in its orbit to compromise between the disturbing perturbations on the constellation caused by Earth's gravity and increased power needed to communicate from increased distance. Because in reality there are gravitational bodies other than the Sun which act on the Sciencecraft, the initial orientation of the triangle has been chosen and the initial states of the Sciencecraft further tweaked in order to minimize the average rate of change of the lengths of the arms of the triangle over the five year operations period of the mission.

Small changes in the geometry of the formation still appreciably effect the phase measurements due to the change in arm length, which varies in relative velocity of between 1 and 15 m/s throughout the year. This also results in changes to the angle between arm lengths of $\pm 1°$. The current baseline design does not take any action to control the relative velocity differences, which results in a doppler effect on the laser beat signals that by the end of the mission moves the signal outside the effective bandwidth of the phasemeter making measurement impossible effectively ending the mission. The change in angle is actively compensated for by articulation between the optic axis defining the arm lengths at a vertex.

2.1.2 INSTRUMENT OVERVIEW

The LISA instrument consists of a constellation of 3 Sciencecraft, each with two proof masses, separated by 5 million km and moving together in an equilateral triangle configuration in orbit around the sun at the same distance as the Earth. The Interferometer Measurement System (IMS) is the part of the LISA instrument that measures the distance between pairs of freely-falling proof masses provided by the Disturbance Reduction System (DRS). The Sciencecraft are all identical, and a single Sciencecraft contains a scientific payload complement of optics and electronics for making the distance measurement and implementing the drag free control, but since the spacecraft itself is an integral part of the DRS, the combination is referred to as a Sciencecraft.

The configuration for LISA interferometry is shown in Figure 2.1-3. Each Sciencecraft has two optical benches, each built around a free-floating proof mass and pointing at one of the other two Sciencecraft. These proof masses form the ends of the interferometer arms.



The distance variations between three Sciencecraft that form the LISA constellation are measured with the Interferometry Measurement System (IMS). The IMS consists of the lasers and optics required to transmit and receive a laser beam between Sciencecraft

Figure 2.1-3: LISA Constellation

2.1.3 Science Data Overview

The basic science data product from LISA is a time series representing the gravitational wave strain sensitivity (h) as a function of time. This time series is generated on the ground by post-processing the raw data streams coming from the instrument (all three Sciencecraft). LISA will provide an absolute measure of the change in distances between the PMs, so no calibration is required *per se*. As there are a number of raw data streams, they may be combined in more than one way, and the set of these combinations is referred to collectively as Time Delay Interferometry (TDI) variables. They are described in more detail below.

Since the instrument is not directional, there will be many signals from all directions added together. Data analysis will be done on the ground with both source-specific and general algorithms to separate these signals using variants of matched filter techniques. The science is developed from the data by extracting the various parameters of each source. The current state of data analysis is described elsewhere[1].

LISA will observe all the sources all the time simultaneously, with scheduled interruptions only for short periods of time needed for communications and maintenance tasks. This eliminates operational constraints, the need of a time allocation for dedicated observations, and prioritization of science objectives.

The data stream relayed to the ground is at an effective continuous data rate of 5 kbps per spacecraft, with approximately 1 kbps of actual displacement measurements, and the remaining 4 kbps a combination of sciencekeeping data and spacecraft housekeeping data. Sciencekeeping data are measurements from auxiliary sensors that provide some context for believing that the science data is acquired under normal conditions. Examples of such data include the charge levels on the proof mass, received power levels on the photodetectors, and an estimate of the residual frequency noise on the laser. This type of information might be used to "veto" the data – i.e determine if there were unusual conditions and that therefore the displacement measurements may be suspect. Housekeeping data are monitors of the overall health of the spacecraft – such as current and voltage of power supplies, temperatures of electronics, and perhaps some measure of the spacecraft attitude control. Details of the data rates and data communications are discussed in Section 3.5.

3 Sciencecraft Design

3.1 OVERVIEW

The Sciencecraft configuration is designed to ensure the requirements of the payload, essentially removal of mechanical and thermal disturbances in the mHz range, are met. The Sciencecraft bus structure consists of a cylindrical exterior wall with a top and bottom panel for direct mounting of the payload and avionics. The primary launch loads are carried through the P/M allowing the bus structure to be constructed from aluminum honeycomb and composites resulting in a relatively light structure. Flexible and deployable appendages will be avoided in order to minimize disturbances in the payload measurement bandwidth, and to eliminate potential failure mechanisms. All inertial sensors are separated as far as possible from other equipment to simplify self-gravity compensation. Thermal stability will be achieved through passive techniques, while active thermal elements (i.e. actively controlled heat pipes, coolers, or louvers) will not be used. A thermally benign payload environment will be achieved through the implementation of a thermally decoupled Solar Array Deck (SAD) large enough to eclipse the Sciencecraft bus. The cylindrical exterior wall and bottom deck will function as radiators to reject electronics waste heat. In addition to housing the payload, the bus structure must also provide mounting accommodations for HGA s and omni antennas for the Comm. System, Coarse Sun Sensors (CSSs) and Star Tracker systems for the ACS, and Micro-Newton thrusters to provide onboard propulsion and fine control for the ACS system. A drawing of the Sciencecraft is shown in Figure 3.1-1. Baseline parameters driving the Sciencecraft design are provided in Table 3-1. A system functional/block diagram is shown in Figure 3.1-2.

Figure 3.1-1: Sciencecraft (side/top view)

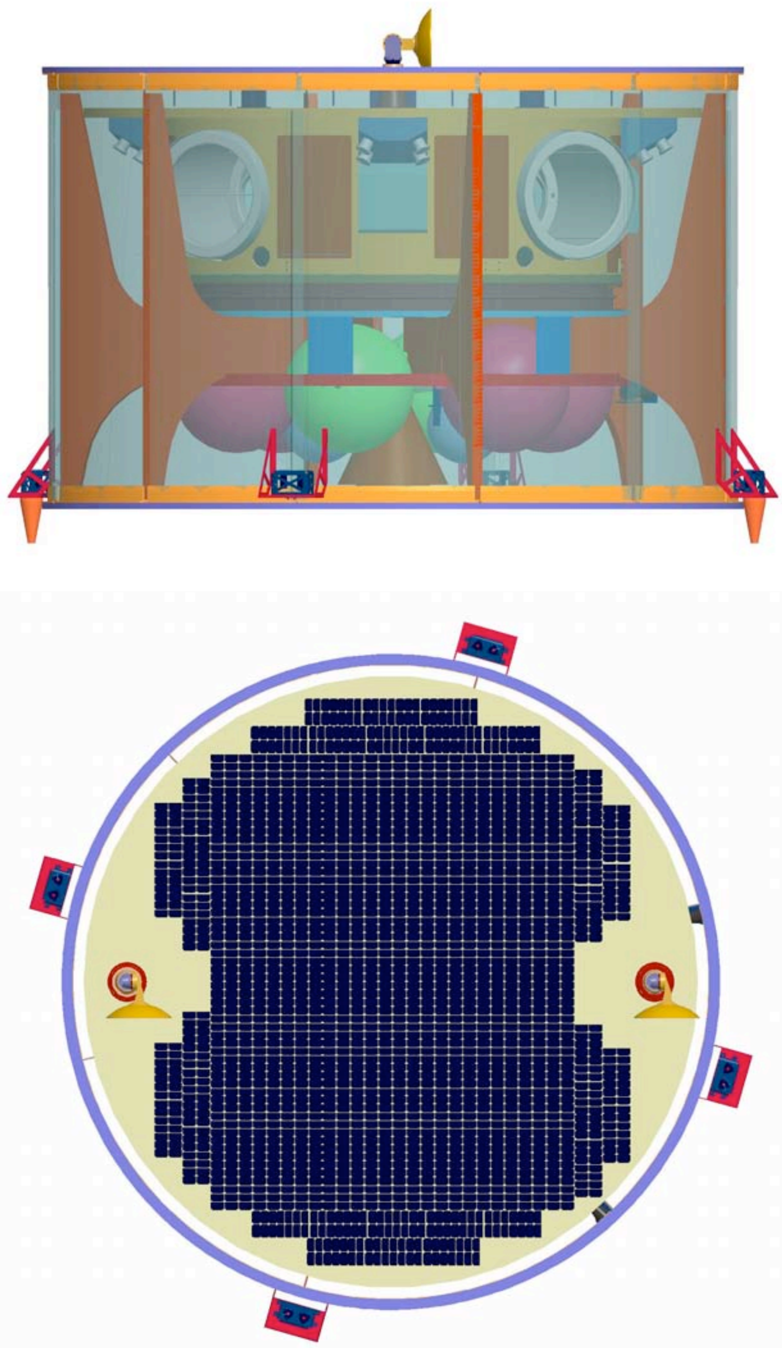
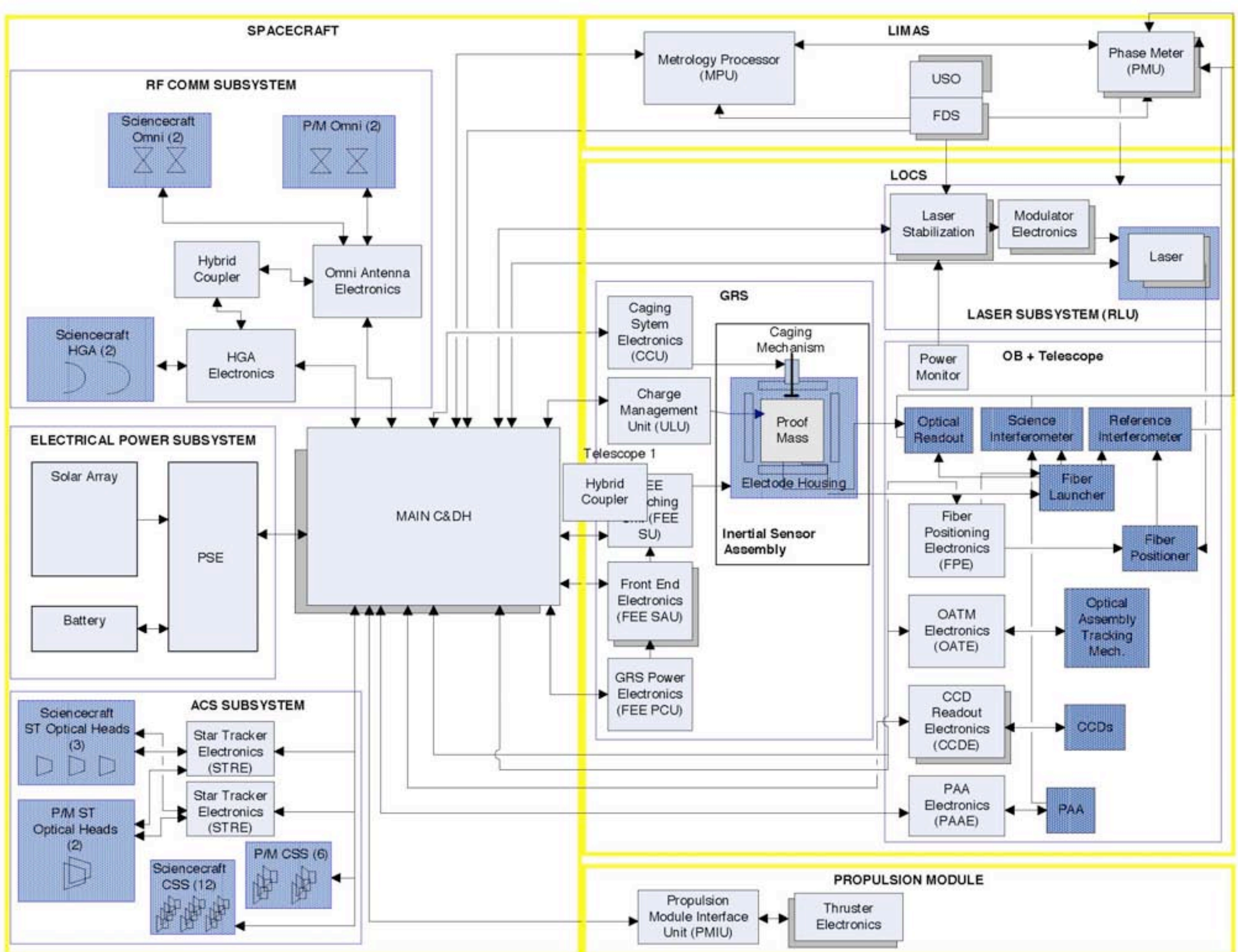


Table 3-1: Sciencecraft Baseline Parameters

Parameter	Value or Definition	Comments
Lifetime	6.5 year minimum 10 year goal	5 year science operations phase after 18 month cruise/commissioning phase
Orbit Transfer Duration	14 months	Must survive launch and orbit transfer within the propulsion module
Commissioning Duration	4 months	
Total Impulse	8300 Ns per thruster	Total propellant required based on 8.5 years of operation (micropropulsion system is not used during transfer)
Power	766 W @ EOL	Electrical power required at end of life
Battery size	1.04 kWh	Covers peak power periods
Thermal Environment	10^{-6} K/√Hz @ 1 mHz Stability (TBR)	Will be achieved using passive methods.
Gravitational Environment	Static grav. field at proof mass < 5×10^{-10} m/s^2 Gravity gradient at proof mass < 3×10^{-8} s^{-2}. Flux distortions at pr. m. < 5×10^{-16} $m/s^2/\sqrt{Hz}$.	Zone sensitive
Magnetic Environment	Magnetic gradient at pr. m. < 5×10^{-6} T/m Magnetic gradient flux at pr. m. < 2.5×10^{-8}	Zone sensitive
Attitude sensing	1 arcsec RMS 3σ (TBR)	Ensure sufficient sensing accuracy during laser beam acquisition phase
Data rate	5 kbps per Sciencecraft 15 kbps for constellation	This includes science data, science housekeeping, and Sciencecraft housekeeping
Communication to Ground	Nominal: Ka-band high-gain Contingency: X-band omni	

Contact scenario	8 hours every 2 days	Minimize the number of rotations fo high-gain antenna (maximize period uninterrupted science data taking)
Interspacecraft Communication	N/A	
Contamination	No thruster plume impingement Outgassing materials must not affect payload and sensitive components	
Reliability	Class B	Single fault tolerant

Figure 3.1-2: Sciencecraft Functional/Block Diagram



3.2 SCIENCECRAFT MECHANICAL SYSTEM DESIGN

Mission baseline parameters driving the LISA Sciencecraft System Design are referenced in Table 3-2.

Table 3-2: Sciencecraft System Baseline Parameters

Parameter	Value or Definition	Comment
Reliability	Class B	Design must be single fault tolerant
Contamination	Thruster plumes must not impinge on or contaminate the payload. Outgassing materials must not affect payload and sensitive components	
AIT	Design must accommodate full accessibility to all components during all Assembly, Integration and Test (AIT) activities.	
Thermal Environment	10^{-6} K/ √Hz at 1 mHz stability	Passive design
Grav.Environment	Static grav. field at PM < $5x10^{-10}$ m/s^2 Grav. gradient at PM < $3x10^{-8}$ s^{-2} Flux distortions at PM < $5x10^{-16}$ m/s^2/√H	Zone sensitive.
Magnetic Environment	Magnetic gradient at proof mass < $5x10^{-6}$ T/m Mag.gradient flux at PM < $2.5x10^{-8}$ T/m/√Hz	Zone sensitive.

3.2.1 MECHANICAL SYSTEM OVERVIEW

The function of the Sciencecraft Mechanical System is to support and protect the payload and Sciencecraft subsystems throughout the entire mission duration and to provide a thermally and dynamically stable environment for the payload during science operations. A bus structure design consisting of honeycomb panel upper and lower decks with aluminum alloy exterior sidewalls and a thermally isolated honeycomb panel Solar Array Deck (SAD) was chosen as the baseline design. A separation system, located on the bottom deck will serve to jettison the Sciencecraft from the Propulsion Module (P/M) during orbital insertion of each Sciencecraft. Each of the three Sciencecraft will be nested inside of a P/M with the three P/Ms stacked into a column inside the launch vehicle payload fairing. The stack design was chosen to carry the launch loads through the Propulsion Module's outer shell, thereby isolating the Sciencecraft from the direct launch load inputs.

The Sciencecraft Mechanical System, consisting of a bus structure and a thermally decoupled Solar Array Deck (SAD), will minimize the use of flexible and/or deployed appendages or launch locks in order to minimize disturbances in the payload measurement bandwidth and to eliminate potential failure mechanisms. All inertial sensors will be separated as far as possible from other payload to simplify self-gravity compensation. Thermal stability will be achieved through passive techniques, while active thermal elements (i.e. actively controlled heat pipes, coolers, or louvers) will not be used. A thermally benign payload environment will be achieved through the implementation of a thermally decoupled Solar Array Deck (SAD) large enough to eclipse the Sciencecraft bus. The SAD will serve two purposes, the first being to provide a substrate for the Electrical Power System (EPS) solar array, and the second being to reject the remaining solar input and/or attenuate the residual thermal inputs to the top deck of the Sciencecraft. The cylindrical exterior wall and bottom deck will function as radiators to reject electronics waste heat. In addition to housing the payload, the Sciencecraft bus structure must also provide mounting accommodations for HGA s and omni antennas for the Comm. System, Coarse Sun Sensors (CSSs) and Star Tracker systems for the ACS, and Micro-Newton thrusters to provide onboard propulsion and fine control for the ACS system.. A dimensioned side view of the Sciencecraft is provided in Figure 3.2-1.

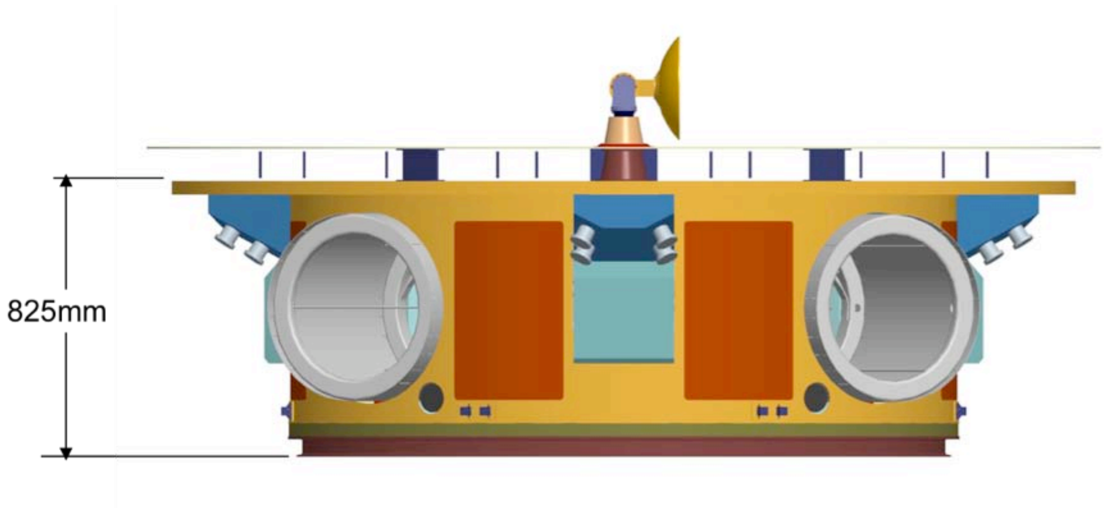


Figure 3.2-1: Sciencecraft Mechanical System Overview

3.2.2 MECHANICAL SYSTEM ARCHITECTURE

A schematic block diagram showing the major components of the Sciencecraft Mechanical System and the major interfaces to the payload and other subsystems is shown in Figure 3.2-2.

3.2.2.1 SCIENCECRAFT BUS STRUCTURE

The Sciencecraft bus structure is designed to accommodate the scientific payload, which consists of the LOCS (LISA Opto-mechanical Core System) assemblies and the LIMAS (LISA Instrument

Metrology and Avionics) boxes. The LOCS assemblies are contained within hexagonal thermal shrouds and incorporate the use of flex-pivots for rotational motion. The flex-pivot assemblies will be integrated to the top and bottom honeycomb decks. The LIMAS boxes and Sciencecraft bus electronics boxes are mounted directly onto the bottom deck panel and the exterior wall. The current design philosophy to isolate the payload and bus electronics from solar heat input and to reject extraneous electrical waste heat into space means that mounting of any electronic components to the top deck panel must be avoided. The Micro-Newton thruster assemblies will attach to the top and bottom deck panels and the side wall sections. Special consideration must be given to the electronics box layout as this can impact the thermal, EMI/EMC and self-gravity environment of the payload. An illustration of the Sciencecraft bus interior showing the LOCS assemblies, the Micro-Newton thrusters, the LIMAS installation and Sciencecraft electronics box layout, and Star Tracker camera head units is provided in Figure 3.2-3. Details of the mechanical and electrical payload to bus interfaces will be captured in separate Interface Control Documents (ICDs).

All initial AIT activities will occur with the solar array and top deck components removed, including the panels themselves. The solar array and top deck components will be installed in the last step(s) of assembly. In order to accommodate removal and replacement of components, maintenance or repairs after the Sciencecraft is completely assembled, access panels will be provided at six locations around the circumference of the bus structure. The Sciencecraft bus structure is illustrated in Figure 3.2-4.

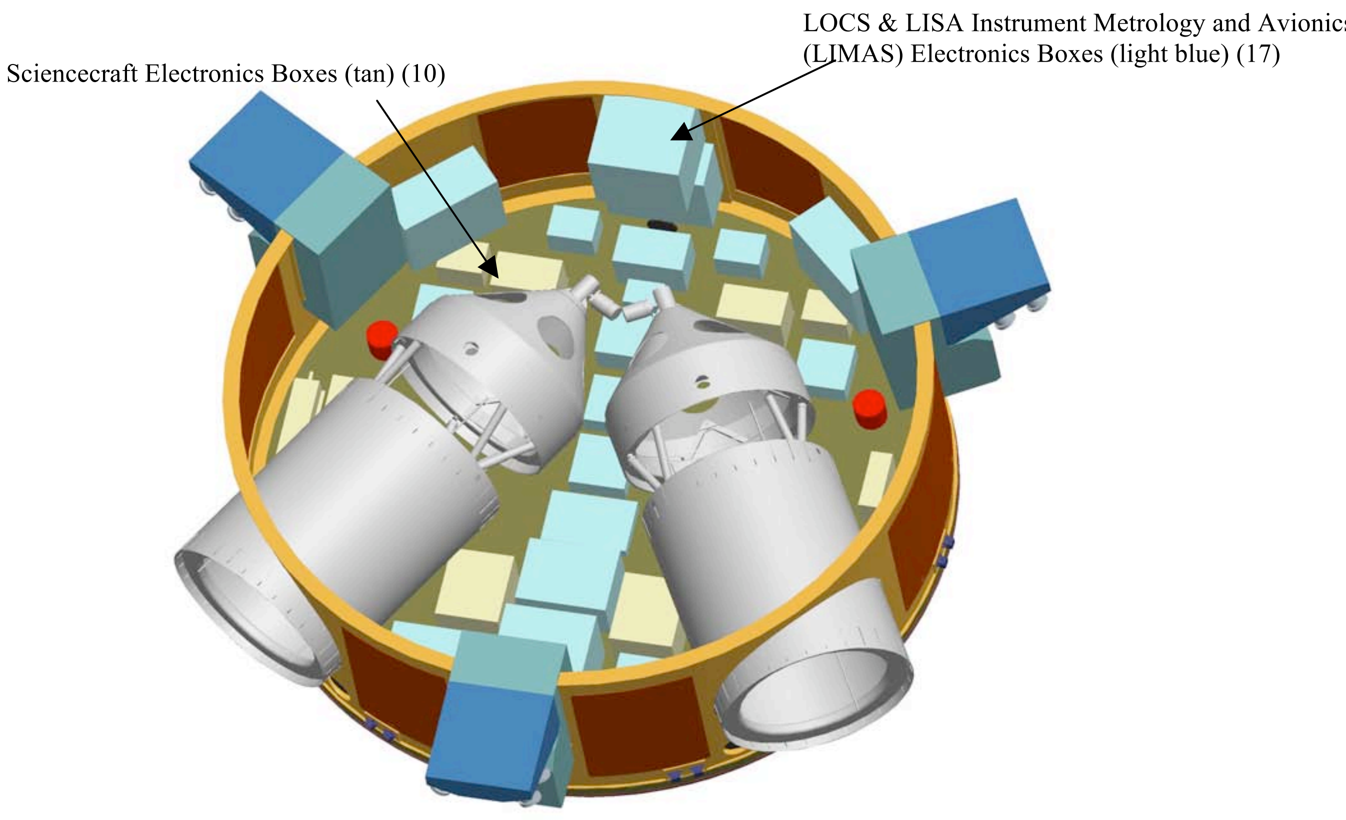


Figure 3.2-3: Sciencecraft Bus Interior

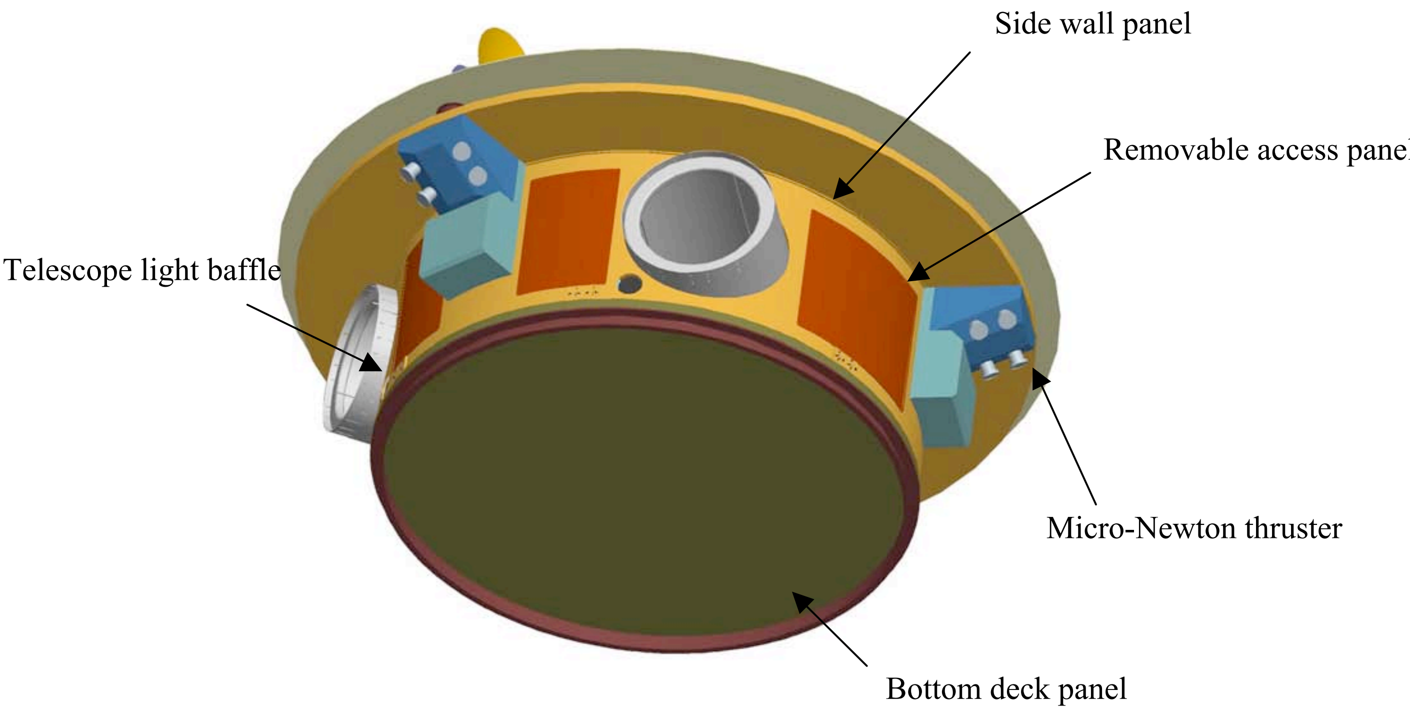


Figure 3.2-4: Sciencecraft Bus Structure

3.2.2.1.1 Sciencecraft Bus Structure Load Path

The Sciencecraft primary load transfer from the solar array and top deck mounted components will be through the cylindrical exterior side wall panels, into the bottom deck panel, through the bottom deck mounted Sciencecraft to P/M separation system and on to the P/M structure as shown in Figure 3.2-5. The loads from all three Sciencecraft modules will be transferred into the P/M stack in this way during launch with the combined load being transferred to the launch vehicle through the Payload Adapter Fitting (PAF).

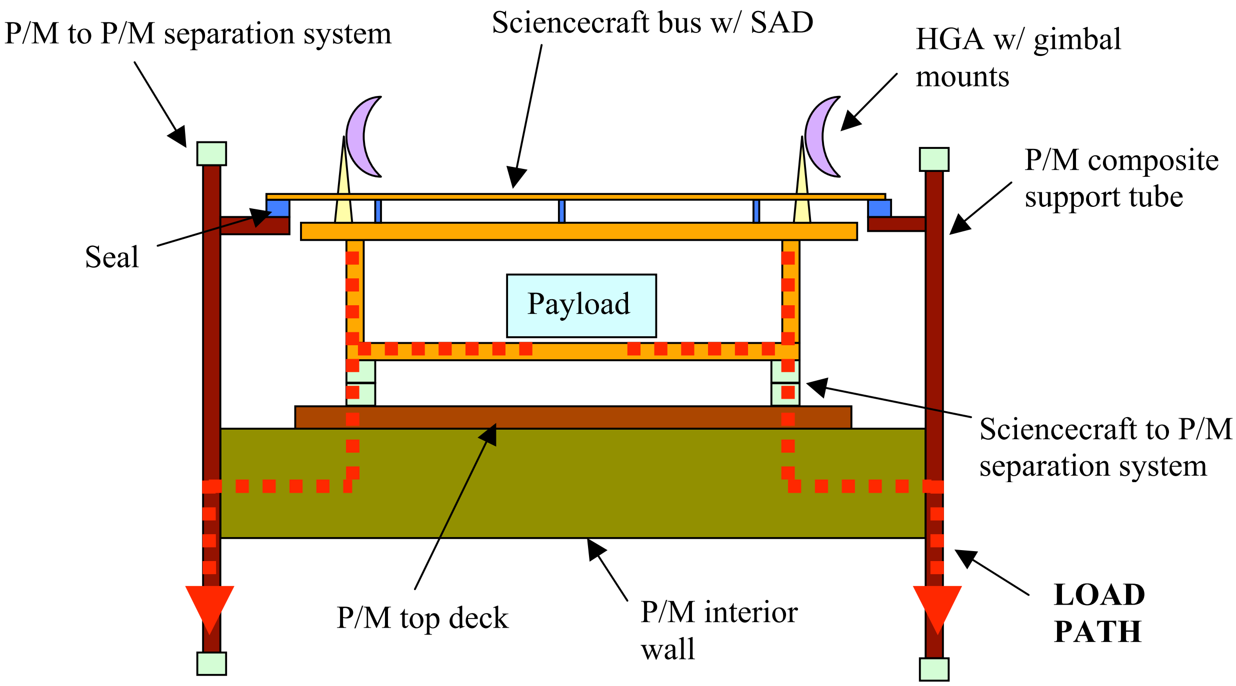


Figure 3.2-5: Sciencecraft Load Path

3.2.2.1.2 Sciencecraft Bus Envelope Dimensions

In order to achieve the most stable thermal environment, the SAD must be large enough to eclipse all components on the Sciencecraft that lie between it and the sun. For this reason the Sciencecraft must fit within a conical envelope due to the 60 degree orientation of the Sciencecraft with respect to the sun line horizontal plane.

Figure 3.2-6 provides dimensions for the Sciencecraft bus as well as for the Solar Array Deck (SAD). The SAD is 2.85 m (9.35 ft) in diameter. The overall diameter of the Sciencecraft bus structure defined by the top deck panel is 2.7 m (8.86 ft). The bottom deck panel diameter is 2.04 m (6.69 ft) and the overall height is 0.585 m (1.92 ft).

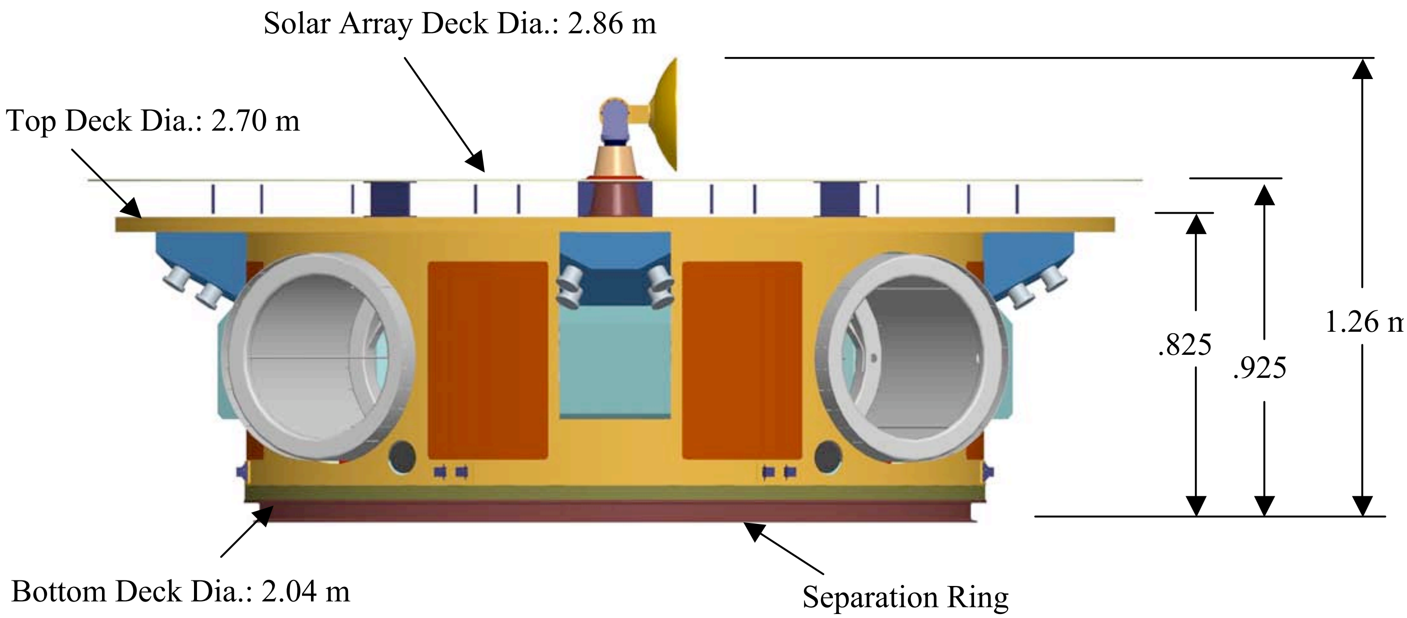


Figure 3.2-6: Sciencecraft Side View

The Sciencecraft envelope dimensions fall within limits driven by the maximum allowed envelope dimensions of the LISA stack. The LISA stack when mounted on the payload adapter fitting must fit within the payload fairing of the objective launch vehicle.

3.2.2.1.3 Material Selection

The top and bottom deck panels will be constructed out of aluminum alloy or composite honeycomb with aluminum face sheets. The cylindrical side wall sections, access panels and telescope interface plates will be made of aluminum alloy.

3.2.2.1.4 Access Panels

Removable panels on the side-walls will allow access to the Sciencecraft interior during AIT activities. Views of the Sciencecraft interior with the access panels removed are shown in Figure 3.2-7.

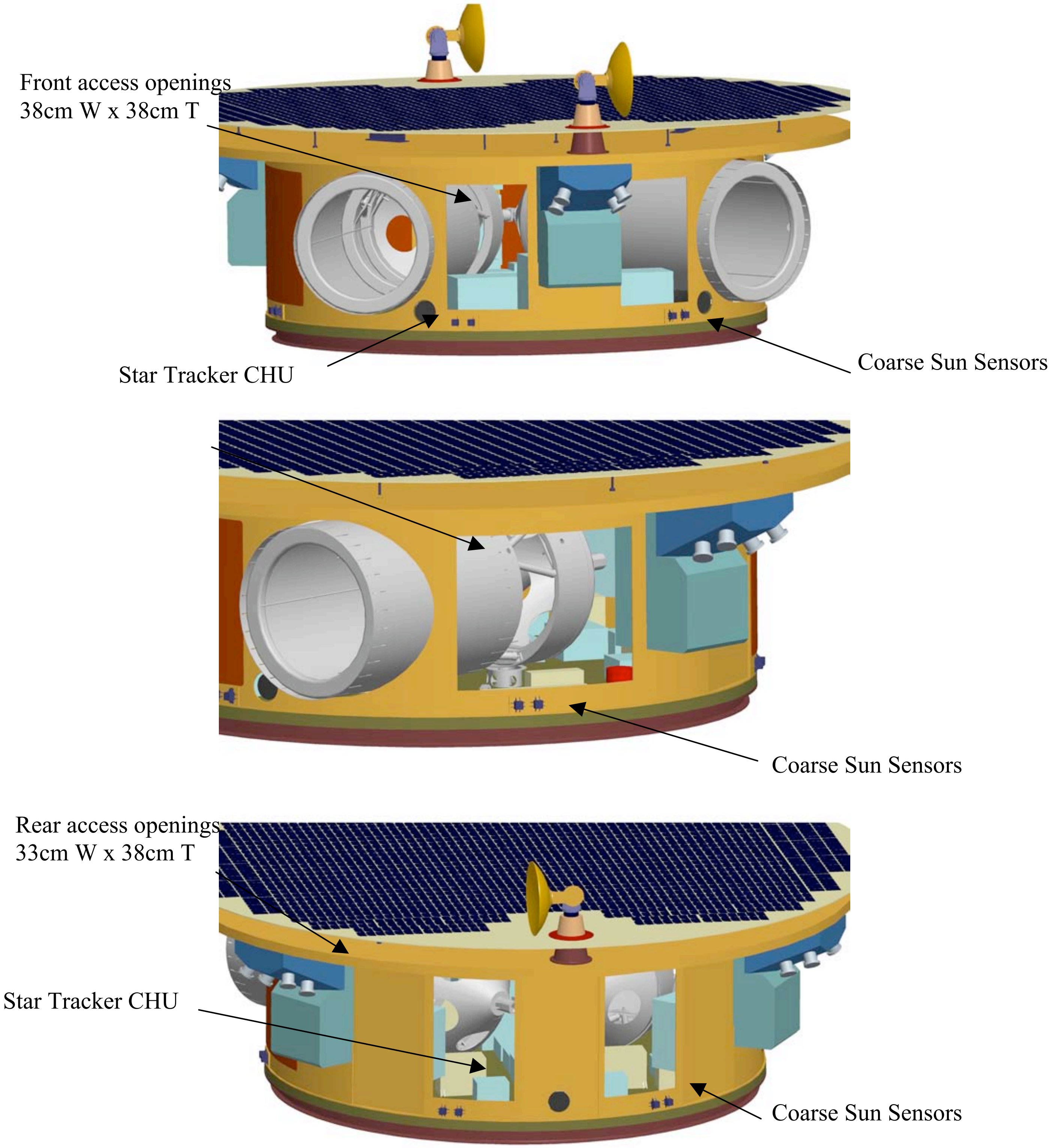


Figure 3.2-7: Sciencecraft Access Openings, ST CHU and CSS Locations

3.2.2.1.5 High Gain Antenna (HGA) Accomodations

Two HGA assemblies will be mounted on the top deck panel through inserts. The HGA gimbal mount units will pass through the SAD. Light baffles will prevent sun light from passing through the gimbal mount cutouts in the SAD. Figure 3.2-8 illustrates the HGA gimbal mount design.

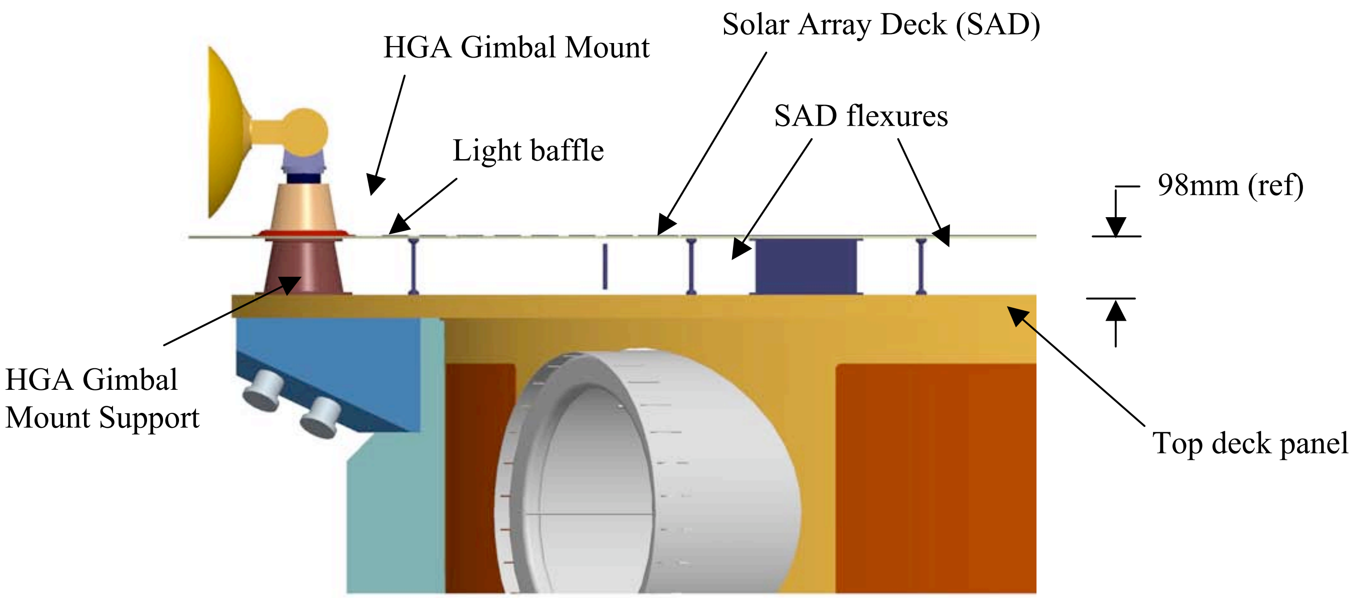


Figure 3.2-8: Top Deck Interfaces

3.2.2.1.6 Star Tracker Camera Head Unit (CHU) Accomodations

The Sciencecraft bus must provide viewing portals for three Star Tracker CHUs. Two of the viewing portals are integrated into removable flanges supporting the telescope light baffles. The third ST CHU is located in the rear of the Sciencecraft bus and is mounted onto the exterior side wall panel. Star Tracker CHU mounting accommodations are shown in Figure 3.2-7.

3.2.2.1.7 Coarse Sun Sensor Accommodations

The Sciencecraft bus must provide mounting provisions for twelve Coarse Sun Sensors (CSSs). The CSSs are configured into two strings of six sensors as a redundant measure. The sensors must be placed around the circumference of the Sciencecraft bus with each sensor string having 360 degrees of sky coverage. The CSSs will attach to the exterior side wall panels as shown in Figure 3.2-7.

3.2.2.2 Solar Array Deck (SAD)

The Solar Array Deck (SAD) provides a substrate for the solar array Photo Voltaic (PV) cells while isolating the Sciencecraft bus from thermal effects of solar radiation. The SAD will be constructed out of a ~4mm thick composite honeycomb panel and will have a diameter of 2.85m (9.35 ft) which is adequate to shield the Sciencecraft bus during science operations. The SAD is dynamically isolated from the Sciencecraft bus structure by a system of I-beam and post flexures. The SAD standoff height from the bus top deck panel is shown in Figure 3.2-8. Figure 3.2-9 shows a layout of the SAD flexures. Figure 3.2-10 shows the Solar Array and HGA layout. The solar array is comprised of 129 strings of PV cells with 14 cells per string. All areas of the Sun Shield not covered by the solar array will be covered by Optical Solar Reflectors (OSRs). Thermal isolation design features of the SAD are discussed in the thermal section of this document. For more information about the solar array refer to the EPS section of this document.

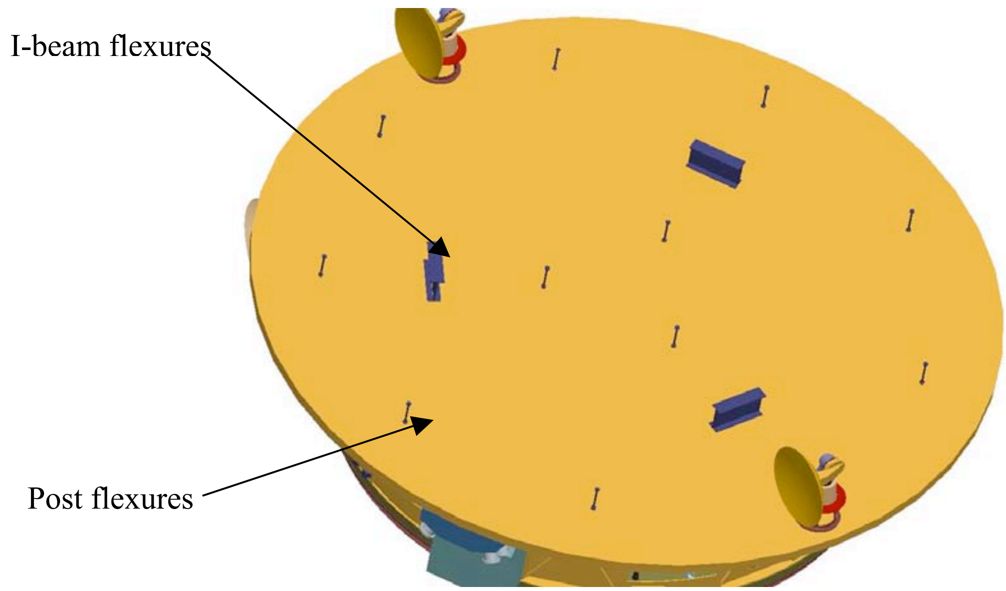


Figure 3.2-9: Solar Array Deck Flexures

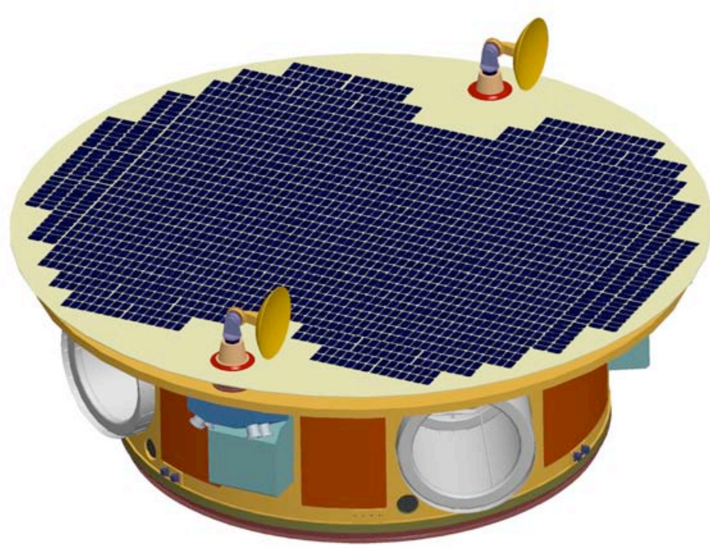


Figure 3.2-10: Solar Array and HGA Layout

3.2.2.3 SEPARATION SYSTEM

The Sciencecraft will be nested inside the Propulsion Module during the Launch and Early Operations (LEOP) and cruise phases of the mission. This configuration will protect the Sciencecraft from orbital debris during LEOP and sunlight exposure on the optical system during sun acquisition maneuvers respectively. In order to achieve a gravitationally balanced system within the given mass budget, it is required that the Sciencecraft separate from the P/M at final orbital insertion. The Sciencecraft will interface with the P/M both mechanically and electrically through a separation system installed on the bottom deck panel of the Sciencecraft bus. A zero insertion force connector built into the separation ring will allow electrical power and control signals to be sent from the Sciencecraft to the P/M electrical components. The Sciencecraft separation ring is shown in Figure 3.2-11.

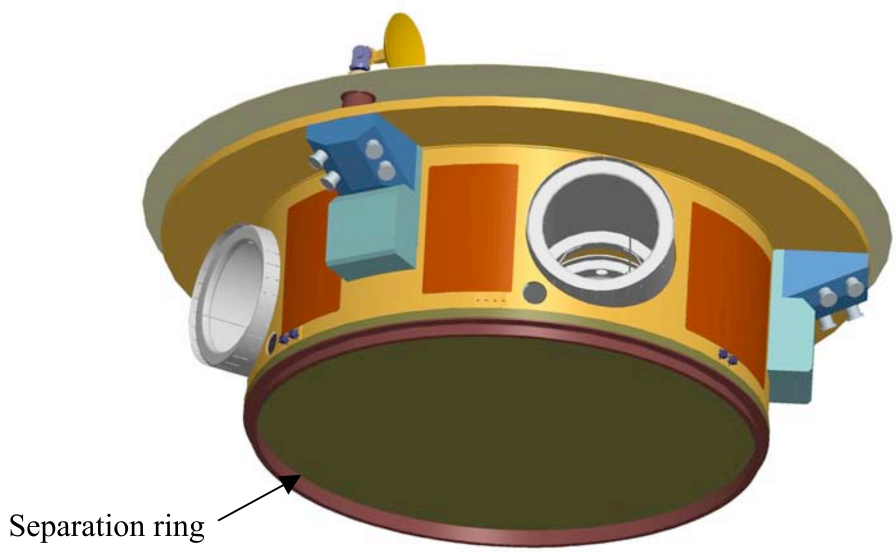


Figure 3.2-11: Sciencecraft Separation System

The baseline design will employ a Motorized Lightband (MLB) separation system made by Planetary Systems Corp. A Lightband system similar to what will be used on the Sciencecraft is shown in Figure 3.2-12. The advantage that the MLB has over conventional separation systems is the fact that the release mechanism uses an electric motor that can be reset quickly while conventional pyrotechnic release systems require a significant amount of time to reset the system. The MLB system will offer a significant advantage during AIT activities. Figure 3.2-13 shows how the Motorized Lightband system works.

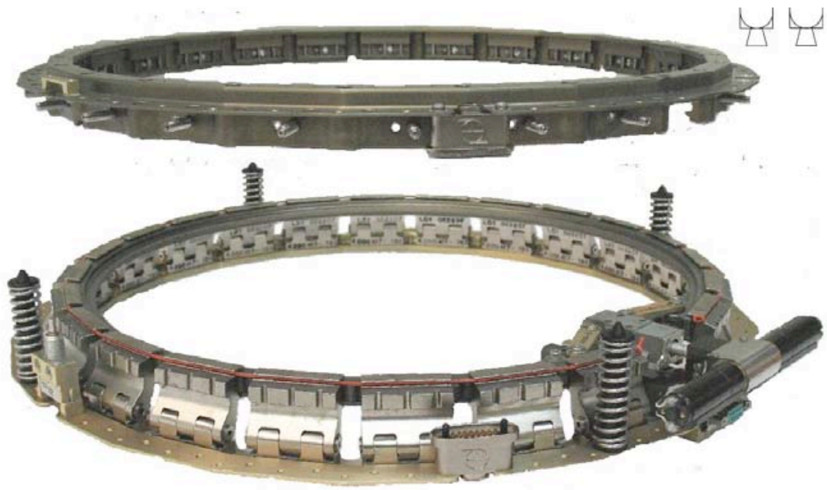


Figure 3.2-12: Motorized Lightband Separation System

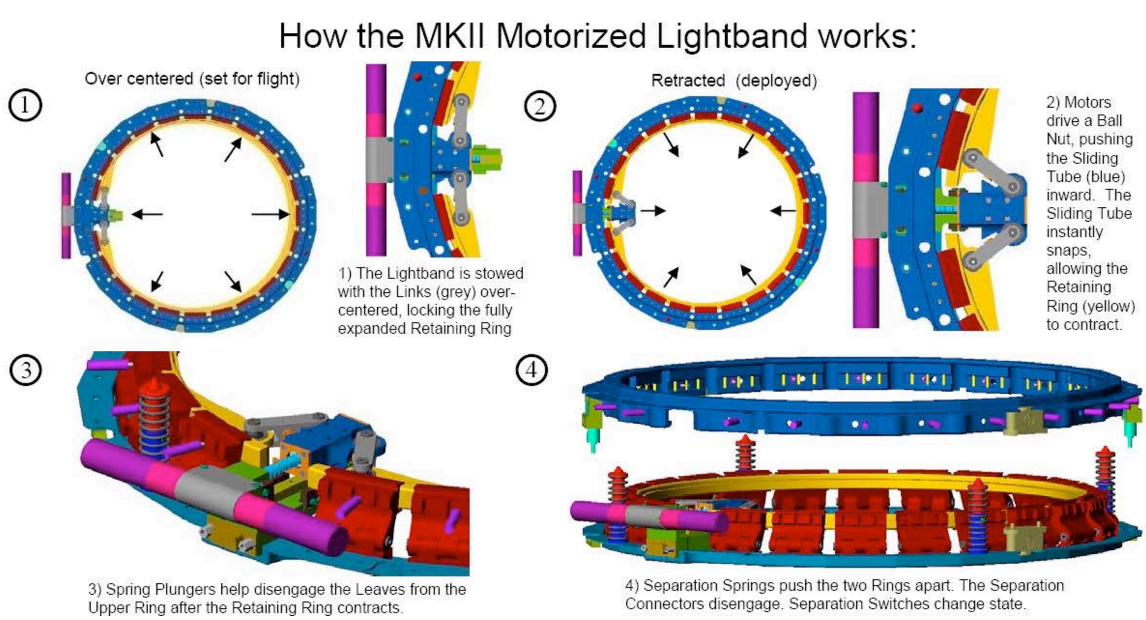


Figure 3.2-13: How The MLB Works

3.2.2.4 ALTERNATIVE SEPARATION SYSTEM

A SAAB Aerospace separation system is an alternative to the baseline Sciencecraft separation system. SAAB separation systems have a record of high reliability with a 100% success rate for more than 350 in-orbit separations between 1981 and 2007. However the release mechanism in the SAAB system relies on pyrotechnic actuation. Resetting the system during AIT activities will likely require replacing the entire pin-puller mechanism which could significantly impact schedule and cost.

3.2.2.5 MICRO-NEWTON THRUSTER ACCOMMODATIONS

The Micro-Newton thrusters are configured as clusters of emitters arranged on the faces of a pyramid. The exact thruster angles will be finalized based on control optimization. The current configuration is taken from the LISA Pathfinder (LPF) geometry, which uses 30° from the local horizontal (i.e. from the mounting surface). Each cluster has an associated electronics module which must be located close to the thruster cluster due to the high voltage (1 kV) circuit required for the thrusters to operate. To minimize the potential for contamination of the telescopes by the thruster plumes, the location and orientation of the thruster emitters is optimized to maximize the separation between the thrust axis and the telescope FOVs. The Micro-Newton thrusters will be mechanically aligned to within 0.5 degrees semi-cone on the ground as part of the AIT activities. At the thruster level, the error between the nominal thrust vector (measured during on-ground test) and the instantaneous thrust direction will be less than 5°. Figure 3.2-14 illustrates the Micro-Newton thruster plumes. The cones represent the cumulative plumes for the emitter clusters. Note that there are no plume impingements upon the Sciencecraft.

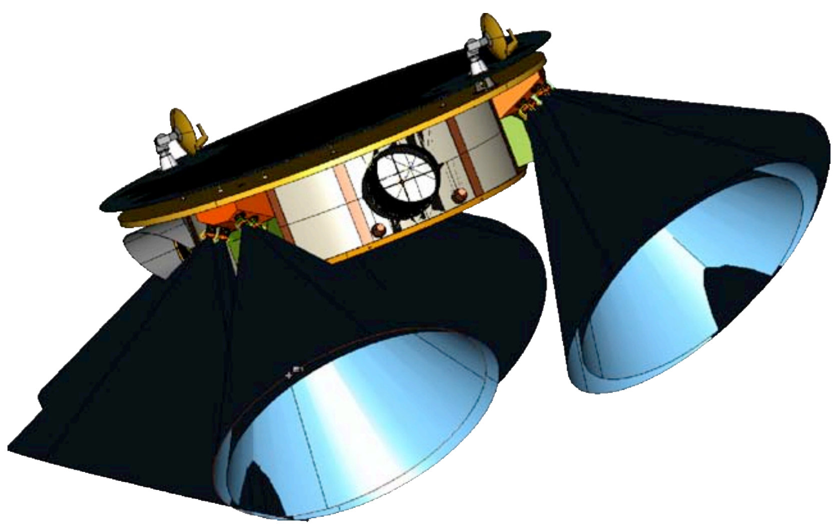


Figure 3.2-14: Micro-Newton Thruster Plumes

3.2.2.6 HGA GIMBAL MOUNT UNITS

The HGAs will be mounted on gimbal units on the top deck of the Sciencecraft bus as shown in Figure 3.2-15. The HGA gimbal mount units will consist of a base component that attaches to the top deck and passes through the SAD, a light baffle to prevent sun light impingement upon the top deck, and two rotating actuators to provide rotation about two axes.

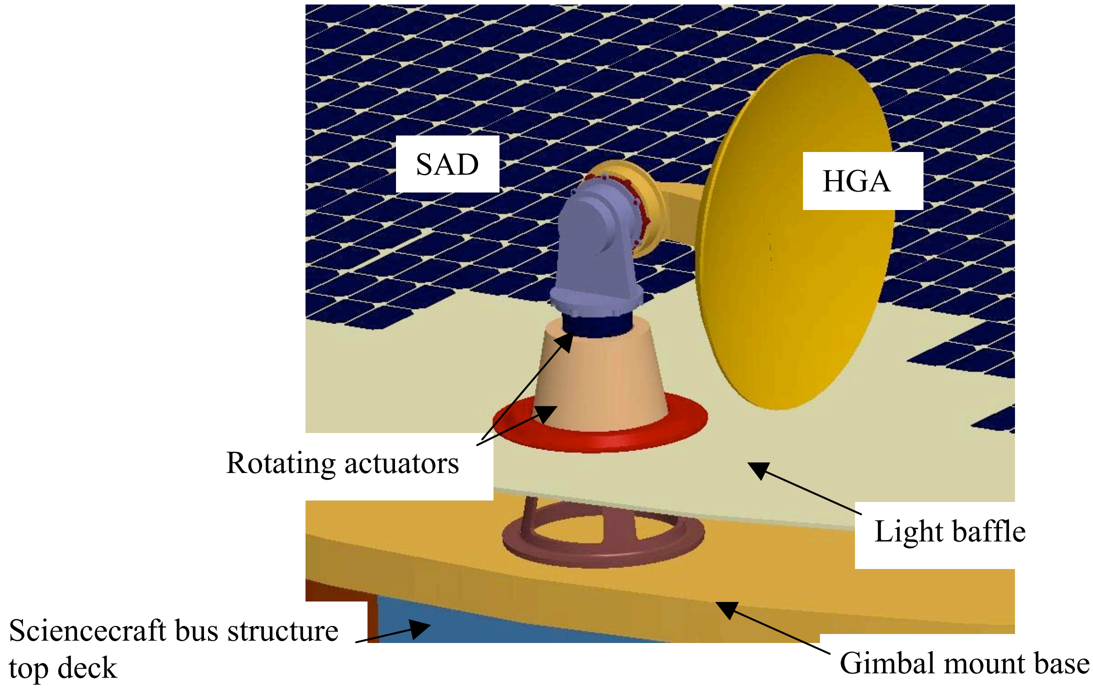


Figure 3.2-15: HGA Gimbal Mount Unit

3.2.3 LISA PAYLOAD ENVIRONMENT

The requirements which are levied upon the proof-mass to obtain a "drag-free" state impose several atypical requirements upon both the LISA mechanical system design and the Sciencecraft system as a whole. These requirements fall under four categories; Self-gravity, Magnetic, Thermal and Structural.

3.2.3.1 SELF-GRAVITY ENVIRONMENT

3.2.3.1.1 Self-Gravity Overview

The force of attraction between two bodies can be expressed by the equation $F = G_0 * m_1 m_2 / r^2$. It is this basic equation that contributes to a unique LISA requirement commonly referred to as "self-gravity balancing". To achieve the acceleration noise budget requirements, the external inputs such as electronics box, actuator, and optical element masses will have to be balanced around the proof-mass. The allocations within the error budget to these disturbances set requirements on the allowable static gravitational field, the gradient of the gravitational field, and the fluctuations of the gravitational field.

The static gravitational field at the proof masses must be kept below 5×10^{-10} m/s^2 along the measurement axes. In meeting this requirement, the amount of static force that must be compensated for by the gravitational reference sensor electrodes will be minimized. This is important for two reasons: first, the force fluctuations generated by the applied compensating electric field are proportional to the total force applied by that field. Fields in both the measurement axes and the other degrees of freedom are important as some of the force fluctuations from other directions will leak into the measurement axis through cross-couplings. Second, the compensating electric field creates a virtual spring between the Sciencecraft and the proof mass. The residual motion of the Sciencecraft will couple through this stiffness causing an acceleration disturbance to the proof mass.

The gradient of the gravitational field at the proof mass locations must be kept below $3x10^{-8}$ s^{-2}. The gravity gradient creates a virtual spring between the Sciencecraft and the proof mass. The residual motion of the Sciencecraft will couple through this stiffness causing an acceleration disturbance to the proof mass.

Finally, fluctuating distortions of the Sciencecraft will change the self-gravity field. These distortions must be minimized such that their acceleration disturbance to the proof masses is kept below $5x10^{-16}$ $m/s^2/\sqrt{Hz}$.

3.2.3.1.2 Self-Gravity Zones

The self-gravity requirements flow down to set requirements on the knowledge of the mass properties and placement of all hardware in the LISA bus. The bus can be divided into zones where all items within a zone have the same knowledge requirements. This does not set the accuracy needed in producing each part; rather it is concerned with the accuracy needed in measuring and identifying the part after it is manufactured. In other words, it defines how well the part must be weighed, measured and placed within the Bus. The current zone definitions and their error allocations are listed in Table 3-3. A detailed discussion of self-gravity zones can be found in reference [2].

Table 3-3: Self-Gravity Zone Definitions and Knowledge Uncertainty

Zone ID	Zone Description	Percent Uncertainty P (%)	Mass Uncertainty, ΔM (kg)	Location Uncertainty $M\,\Delta R$ (kg-m)	Dimensional Uncertainty ΔL (m)	Other Allowable Uncertainties
A	Proof masses	1	0.0254	-	4.2e-5	0.68 % variation in density, 0.173 mm skew, 0.134 mm taper
B	Proof mass housings	4	1.5e-5	8.7e-7	0.00135	0.132 mm skew
C	Optical bench	4	7.2e-4	1.65e-6 and/or 4.1e-5 M	5.8e-5	0.087 deg. overall rotation
D	Telescope assembly	1	0.0037	4.1e-4	-	
E	Payload tube outer surface	1	0.0038	4.3e-4 and/or 5.4e-4 M	2.4e-4	Generic formula: $\Delta M < 0.0749\ P\ R^2$ $M\,\Delta R < 0.0375\ \ P\ R^3$
F	Outside the payload tube, beyond 250 mm of a proof mass	0.5	0.0023	2.9e-4	-	
G	Outside the payload tube, beyond 500 mm of a proof mass	0.5	0.0094	0.0023	-	
H	High Gain Antennas	0.5	0.033	0.0038	-	
I	Solar array	0.5	0.043	30 x 0.00163	-	

3.2.3.2 MAGNETIC ENVIRONMENT

The magnetic properties of the proof mass and characteristics of the magnetic field contribute to several effects throughout the DRS error budget. The leading effects are: the interaction between the fluctuating interplanetary magnetic field with the S/C magnetic gradient, fluctuations from the magnetic gradient induced current dissipations (Eddy current damping), and a fluctuating S/C magnetic gradient. The magnetic field inside the spacecraft is driven by the magnetically hot components. The current budget value for the magnetic gradient at the proof mass is 5×10^{-6} T/m, which is equivalent to an 8.3 A-m^2 maximally oriented dipole one meter away. Magnetic gradient fluctuations must be kept below

2.5×10^{-8} T/m/√Hz at 0.1 mHz at the proof mass locations. Magnetic parts are generally avoided in the LISA design, however it is not cost effective to completely eliminate them all. All magnetic items are tracked within System Engineering so that a magnetic budget can be maintained. The budget magnetic field should be easily met by placing all the magnetic components far from the proof mass and including a modest amount of magnetic shielding or compensation. While care must be taken in controlling the magnetic field within the LISA S/C, its requirements on the magnetic field are not as challenging as true "magnetically clean" spacecraft such as those used to measure the interplanetary magnetic field.

The layout of the bus and the payload contribute to the overall magnetic field that affects the magnetic properties of the proof mass, i.e. contributes to the noise. In order to estimate the magnetic field performance, the magnetic properties of all the components must be tracked. Initial tracking includes the location of the component in the S/C and an estimate of its dipole moment. Eventually, higher order magnetic moments and magnetic fluctuations will also be tracked[3].

Table 3-4 lists the currently identified magnetically hot items within the systems currently being tracked in the detailed mass budget. While this list is not complete, the two items with estimates are expected to be the strongest permanent magnets on-board the Sciencecraft. The current gradient requirement is 5×10^{-6} T/m.

Table 3-4: Current List of Magnetically Hot Items

Component	Quantity	Dipole (A-m^2)	Reference
HGA Drive Mechanism	2		
Transponders	2		
RFDU	1		
Heaters	Many		
Solar Array	1		
Battery (9A/h LiIon)	1		
Power System Electronics	1		
Power Switching & Distribution Unit (PSDU)	1		
SSPA/TWTA	2	15	Cassini
Lasers	4		
Isolator	4	10	Optics for Research (OFR) fixed isolator

Very high permeability foil magnetic materials like METGLAS (Metglas Solutions, Inc) and VITROVAC (Vacuumschmeltze) allow easy shielding of hot items to reduce their magnetic signature. The significant advantage of these materials is that they can be easily formed in place and cold working does not reduce their permeability. Shielding should be used sparingly and only after other means of eliminating or reducing the magnetic signature have been exhausted or deemed not to be cost effective. The most effective use of magnetic shielding is to contain the large, permanent fields associated with relays, electromagnets, stepper motors and other time variable fields. The shielding material provides a "shunt" for the magnetic field lines so they return to the poles through a low reluctance path rather than free space. The shielding material should be used close to the source and possibly integrated into the structure of the device itself[4].

In order to control the contributions to magnetic noise a preliminary set of zones has been established as shown in Table 3-5. These zones influence the placement of all bus and payload components.

Table 3-5: Preliminary Magnetic Zones

Zone	Name	Minimum Distance to Nearest PM (mm)	Maximum Magnetic Dipole ($A\text{-}m^2$)
A	Proof masses	0	0
B	Proof mass housings	2	2.7e-12
C	Optical bench	44	6.2e-7
D	Telescope assembly	222	4.0e-4
E	Y-tube outer surface	225	4.3e-4
F	Outside the Y-tube, beyond 250 mm	250	6.5e-4
G	Outside the Y-tube, beyond 500 mm	500	1.0e-2
H	High Gain Antennas	936	1.3e-1
I	Solar array	200	2.7e-4

3.2.3.3 THERMAL ENVIRONMENT

All elements are thermally isolated in order to secure thermoelastic stability. To further ensure disturbance minimization, an extremely stable thermal environment is required, with no active thermal elements able to induce mHz disturbances at the payload interface. Effectively this requirement drives the payload thermal environment to be well decoupled from both solar radiation and in turn from the SC structure itself.

3.2.4 ANALYSIS

3.2.4.1 STRUCTURAL THERMAL OPTICAL & GRAVITATIONAL (STOP-G) ANALYSIS

3.2.4.1.1 STOP-G Analysis Overview

A full system measurement of the Sciencecraft self-gravity is most likely not practical to these levels. LISA will rely on mass property and position measurements combined with modeling to verify the system meets the self-gravity requirements. The LISA integrated modeling team developed a structural, thermal, optical, and gravitational analysis method (STOP-G) for the purpose of verifying the self-gravity requirements both during the design process and Sciencecraft integration and test[5]. The STOP-G process begins with a geometric, solid representation of the Sciencecraft that is used to produce a single Finite Element Model (FEM). The FEM is then passed to the thermal engineer to generate temperature predictions. The nodal temperature results are then passed to the structural engineer to determine the thermal distortions as a result of thermal effects. These distortions are then passed in parallel to optics and self-gravity. The optics engineer uses the distortions to determine if any unacceptable misalignment and displacements of optical components occur. The gravity engineer determines the self-gravity effects and evaluates if the forces and gradients are within acceptable parameters. A custom code was written to perform the self-gravity analysis. The tool requires as input the FEM mass matrix, a set of rigid body vectors, and the deformed Sciencecraft node locations. It uses the FEM for nodal mass and location definitions, and calculates the self-gravity forces, moments and gradients on each proof mass according to Newton's law of gravitation for a total of 3 forces, 3 moments and 36 gradients on each proof mass. A point mass approximation is used to calculate the gravitational forces and moments. Superposition makes the calculations separable and adding all the contributions trivial. The use of a point mass approximation means imposing strict requirements on the grid sizes used in the model to achieve the required accuracy. The most recent end-to-end self gravity analysis of the LISA design can be found in references [6] and [7]. A detailed description of the self-gravity tool can be found in reference [8].

A complete self-gravity analysis consists of the following parts:

Static self-gravity analysis – calculate all self-gravity forces, moments, and gradients for the nominal configuration of the S/C.

Thermal deformation - calculate the deformation of the Sciencecraft due to the ground-to-orbit temperature changes and the resulting change in self-gravity

Moving parts – calculate the change in self-gravity due to the repositioning of any moving parts on the S/C including optical assembly articulation, high gain antenna rotation, and thruster propellant use.

3.3 ELECTRICAL POWER SYSTEM DESIGN

Mission baseline parameters driving the LISA Electrical Power System (EPS) design are referenced in Table 3-6.

Table 3-6: Electrical Power System Baseline Parameters

Parameter	Requirement	Comments
Lifetime	+ 5 year science operations phase (10 year goal) after 18 month cruise phase	14 months transfer trajectory +4 commissioning
Orbit	3 independent, Heliocentric, 20° earth trailing orbits, equilateral triangular constellation with 5 x 10^6 km +/- 1% arm lengths. 30° incident angle between bus normal and solar illumination vectors.	
Thermal Stability @payload interface	10^{-6} K/√Hz @ 1 mHz TBR	Power subsystem must not contribute excessively to the thermal noise at the payload interface
Battery Lifetime	+ 5 year science operations phase (10 yr goal) after 18 month cruise phase	Designed to meet peak power requirement at EOL
Bus voltage	28 V ±0.14 V.	Voltage supply to the payload, Bus voltage regulation will be 28 V ±2 V (TBR)

3.3.1 Electrical Power System Overview

The function of the Electrical Power System (EPS) consists of power generation, regulated power distribution to support the electrical needs of the payload and other subsystems and Command and Data Handling (C&DH) for the payload and all S/C subsystems. The primary constituents of the EPS are the Solar Array (SA) to provide power generation, a low Ah battery used primarily during the Launch and Early Operation (LEOP) Phase and a Maximum Peak Power Tracker (MPPT) to regulate the power supply.

In addition to providing for Sciencecraft electrical power needs, the EPS also provides power for the P/M via an umbilical connection across the separation interface plane to operate propulsion system valves, pressure transducers, survival heaters, Star Trackers, Coarse Sun Sensors and omni antennas.

The EPS must operate with low thermal fluctuation and low electromagnetic field generation to provide a low noise environment for the payload during science observations.

3.3.2 Electrical Power System Architecture

The primary constituents of the EPS, as shown in system functional block diagram (Figure 3.3-2), consists of the SA, the Li-Ion battery and the power control and distribution system. A standard electrical data bus such as MIL-STD-1553 or Spacewire will be used for all command and data handling functions. Electrical power will be distributed via standard harnessing, with the potential use of flex circuitry if it is required to mitigate self-gravity contributions.

3.3.2.1 SOLAR ARRAY

The solar array will provide all of the power for the Sciencecraft during normal science observations. The solar array will also provide power to the Sciencecraft and the P/M during the cruise phase. S/C design and orbit transfer maneuvers will be constrained to ensure that the solar array remains illuminated throughout the cruise phase.

3.3.2.1.1 Solar Array Deck Mounting

The solar array will be mounted on the Solar Array Deck (SAD). The SAD also functions as a sun shield that is thermally isolated from the Sciencecraft bus structure. The sun shield design is necessary to ensure the thermal stability required during science observations. An in-depth description of the sun shield thermal design is provided in Section 5.7. A fixed mounted solar array was selected both to avoid uncertainty in mass location that could result in acceleration noise, and to eliminate the risk of deployment mechanism complications. The solar array design is illustrated in Figure 3.3-2.

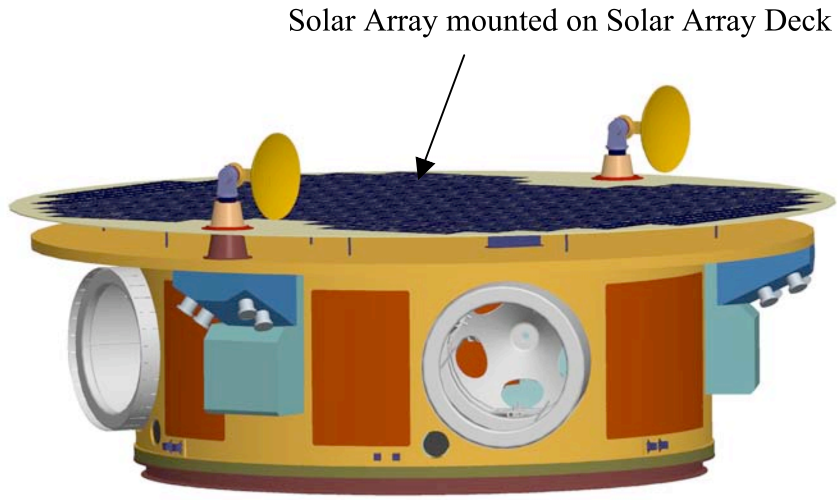


Figure 3.3-2 Solar Array Mounting

3.3.2.1.2 Solar Cell String Design

The solar array cells will be configured into 129 strings with each string containing 14 standard triple junction Gallium Arsenide (TJGaAs) cells. The 14 cell string size balances voltage and current to provide the optimal power generation per unit area. This design assumes that full battery charge at EOL is not required. With a total of 1806 solar cells measuring 6cm x 4cm each, the effective solar array area is 4.33m^2. The standard solar cell strings will be arranged in a 2x7 cell formation. Assuming a solar cell packing efficiency of 82%, the required surface area for the solar array is estimated to be approximately 5.3m^2. The 129 solar cell string count includes 4 additional strings for increased reliability.

The solar array cell layout is shown in Figure 3.3-3. The layout accounts for shadowing from the two HGAs, and ensures no SA shadowing will occur during the cruise and science mission phases. Strings with non-standard cell arrangements will be placed around the periphery of the array.

A silver mesh sheet will be applied in a nearly cell-wide strip underneath each string. The mesh between substrings will be connected using the mesh or with wire, with the negative wires twisted together with the positive wires or simply adjacent if there is not enough length to twist. The substrate surface will have a Kapton insulator. The mesh will get laid onto that with an adhesive. Another insulating layer will be laid on top of the mesh. The cell string or substring will then be laid on top of this insulator with the same adhesive.

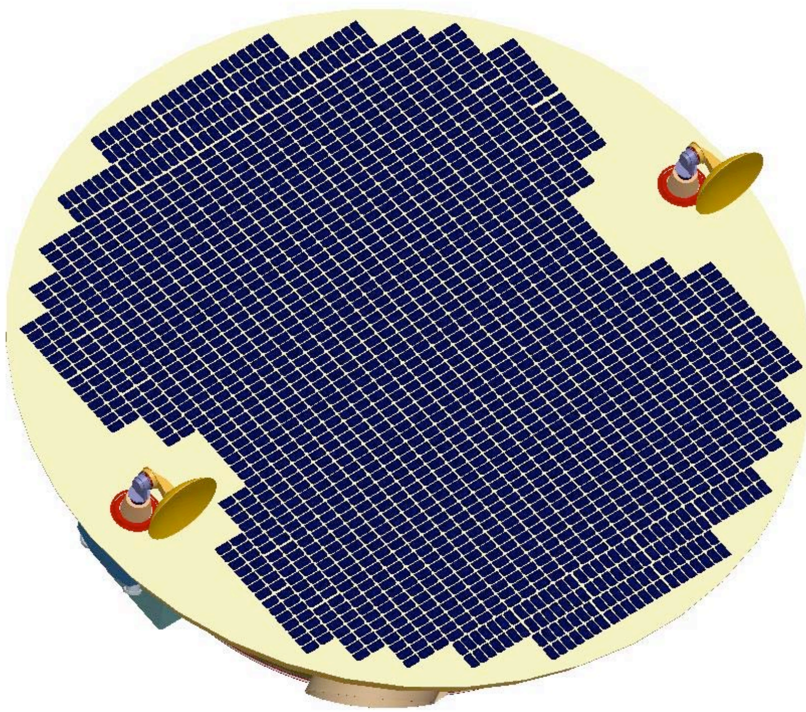


Figure 3.3-3: Solar Array Layout

3.3.2.1.3 Solar Array Sizing

Solar array sizing is based on the S/C electrical load analysis provided in Table 3-7 and the solar array sizing parameters provided in Table 3-8. The electrical load analysis provides estimated electrical loads for each S/C subsystem for each mission phase with a 30% contingency accounted for each load. The analysis shows a maximum power requirement of 794W occurring during normal science operations. The solar array will require battery assistance to meet the estimated payload peak power requirement of 1.1 kW.

Orbital mechanics of the mission require the Sciencecraft normal vector, and hence the solar array normal vector, to be oriented at a fixed 30 degree angle of incidence to the sun during science observations. For this reason, solar array sizing must account for a 30 degree cosine factor.

Table 3-7: Spacecraft Electrical Load Analysis

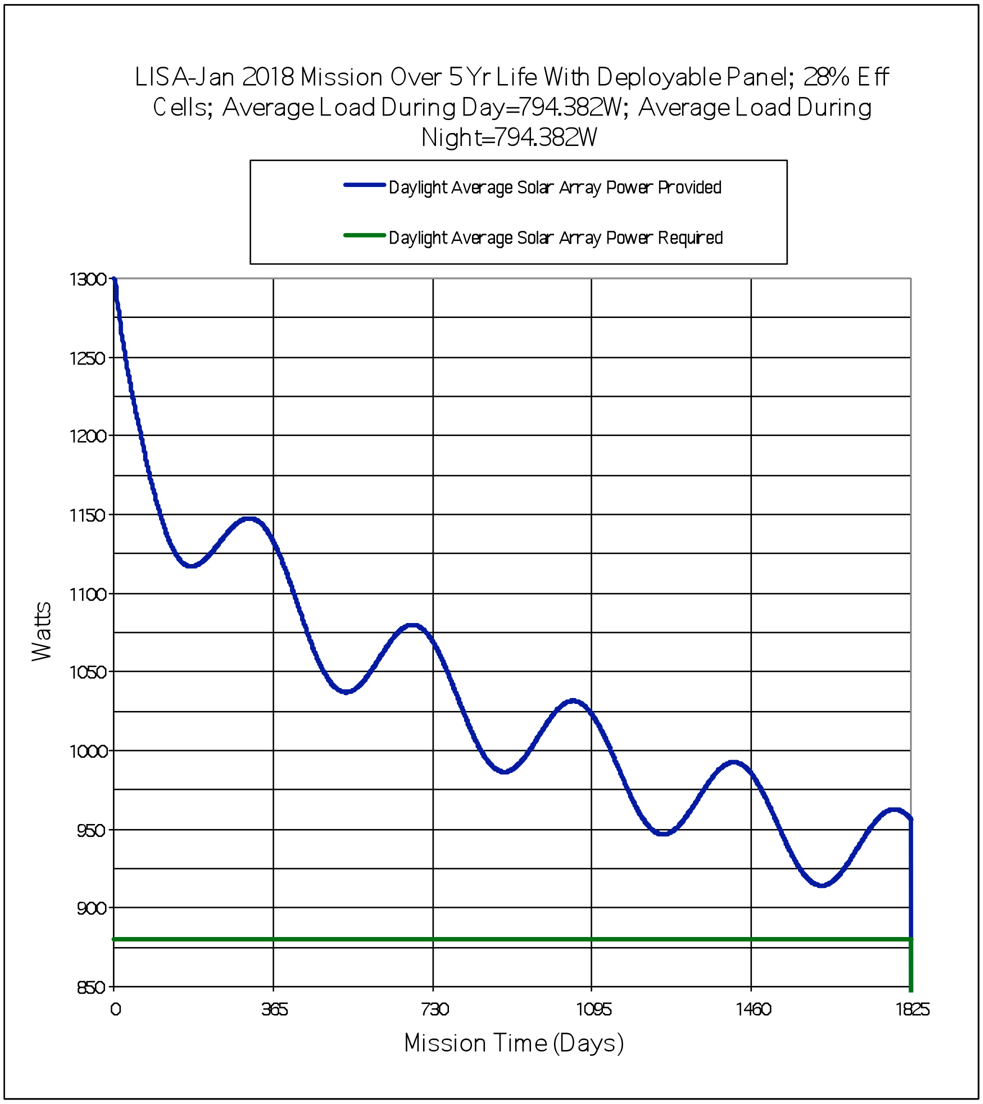
5 year Missinon Life				Science Operations	Commission Power	Safe Hold				
EPS Load Item Description				Avg. Power Watts	Avg. Power Watts	Power in Watts	Peak Power	Launch Power Requirement	Cruise Power	Comm Downlink Power
Total Power				794.4	794.4	490.8	1,102.5	129.0	383.6	745.8
Time Period Over Which Averaging Is Done For Each Mode (min.)		Contingency %								
	Inst Global Contingency	30								
Instruments with Contingency				328.5	328.5	65.7	411.3	39.0	65.0	328.5
LOCS				173.0	173.0	34.6	196.8	0.0	0	173.0
	Contingency	30		51.9	51.9	10.4	59.0	0.0	0.0	51.9
LIMAS				79.7	79.7	15.9	119.6	0.0	0	79.7
	Contingency	30		23.9	23.9	4.8	35.9	0.0	0.0	23.9
Heaters				0.0	0.0	0.0	0.0	30.0	50.0	0.0
	Contingency	30		0.0	0.0	0.0	0.0	9.0	15.0	0.0
Spacecraft Loads with Contingency				465.9	465.9	425.1	691.2	90.0	318.6	417.2
	Spcft Global Contingency	30								
PSE	MAP like (95.4% eff)	36.54		37.0	37.0	22.6	51.0	5.9	17.5	35.0
	Contingency	30		11.1	11.1	6.8	15.3	1.8	5.3	10.5
Electrical - Harness Losses				7.6	7.6	4.7	10.6	1.2	3.6	7.2
	Contingency	30		2.3	2.3	1.4	3.2	0.4	1.1	2.2
Calculated				63.0	63.0	24.0	133.0	24.0	63.0	63.0
	Contingency	30		18.9	18.9	7.2	39.9	7.2	18.9	18.9
Solid State Data Recorder				0.0	0.0	0.0	0.0	0.0	0.0	0.0
	Contingency	30		0.0	0.0	0.0	0.0	0.0	0.0	0.0
Solar Array Drive Motor				0.0	0.0	0.0	0.0	0.0	0.0	0.0
	Contingency	30		0.0	0.0	0.0	0.0	0.0	0.0	0.0
Solar Array Drive Electronics				0.0	0.0	0.0	21.4	0.0	0.0	0.0
	Contingency	30		0.0	0.0	0.0	6.4	0.0	0.0	0.0
Attitude Control				29.0	29.0	29.0	29.0	0.0	12.0	29.0
	Contingency	30		8.7	8.7	8.7	8.7	0.0	3.6	8.7
Thermal				35.0	35.0	60.0	100.0	35.0	100.0	0.0
	Contingency	30		10.5	10.5	18.0	30.0	10.5	30.0	0.0
Propulsion				127.2	127.2	127.2	127.2	0.0	9.0	127.2
	Contingency	30		4.0	4.0	4.0	4.0	4.0	2.7	4.0
Data Systems				85.8	85.8	85.8	85.8	0.0	40.0	85.8
	Contingency	30		25.7	25.7	25.7	25.7	0.0	12.0	25.7
Com, X Band Transmeter				9.7	9.7	1.9	14.6	1.9	1.9	9.7
	Contingency	30		2.9	2.9	0.6	4.4	0.6	0.6	2.9
Com, S Band Transmeter				9.0	9.0	1.8	13.5	1.8	1.8	9.0
	Contingency	30		2.7	2.7	0.5	4.1	0.5	0.5	2.7
Com, Reciever				8.0	8.0	1.6	12.0	1.6	1.6	8.0
	Contingency	30		2.4	2.4	0.5	3.6	0.5	0.5	2.4

Source: MDL January 2008

Table 3-8: Solar Array Sizing Parameters

Parameter	Value	Comment
Science mode power req't	794.4 W	
Solar cell efficiency	28%	
Power fluence	2.361E+14	Used to characterize degradation due to radiation. Assume 6.5 yr 1 MeV equivalent.
Nominal Bus Voltage	28 V	
30 degree cosine factor	.866	30 degree sun incidence angle
Operating Temperature	86 degrees C	
String Size	14 cells in series	
Cell size	6cm x 4cm	
Reliability design	4 add'l strings	

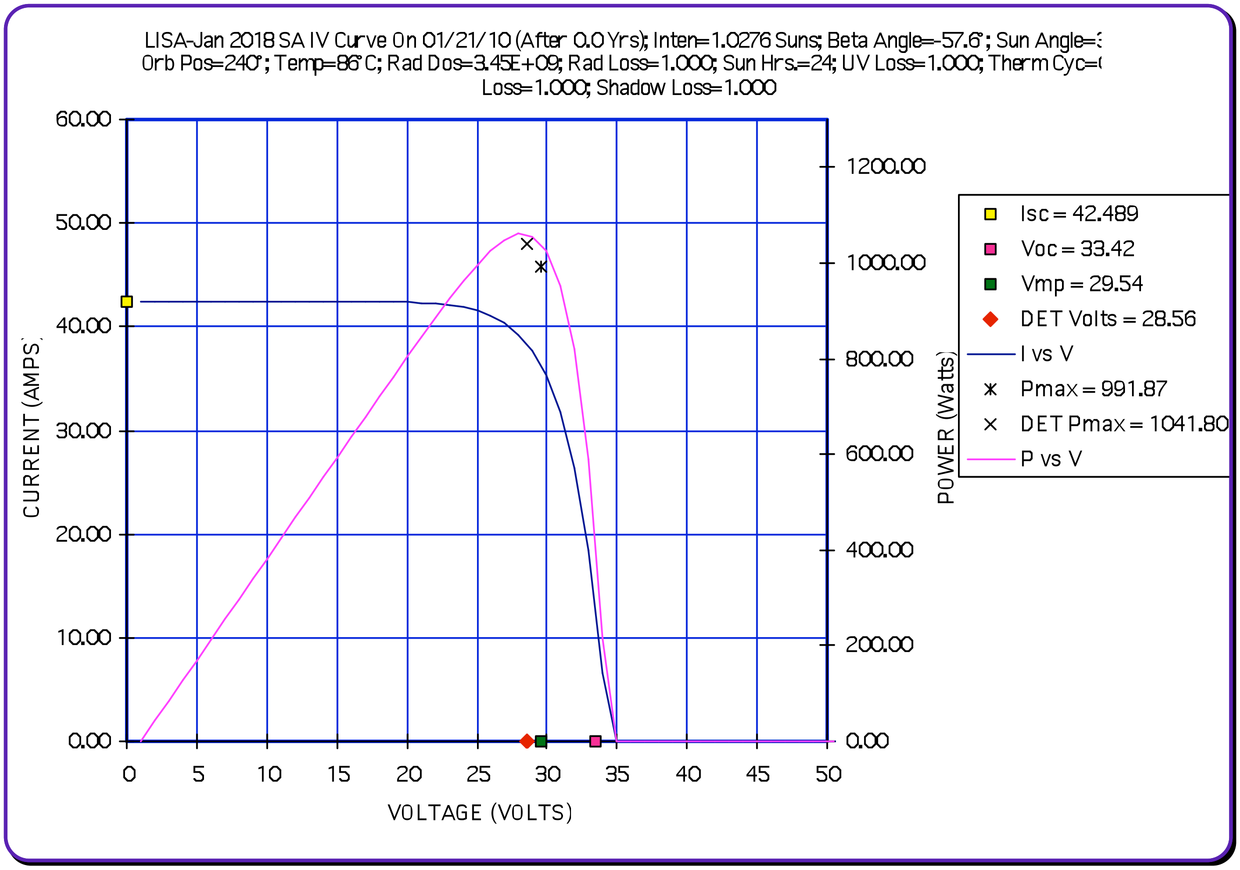
Figure 3.3-4 illustrates the solar array electrical power curve throughout the five year science mission duration compared to the constant electrical load requirement. Annual solar radiation flux and lifetime degradation are evident in the plot. During the last year of the mission, the power provided by the solar array will bottom out at approximately 915W, exceeding the 794W power requirement by 15%.



Source: MDL January 2008

Figure 3.3-4: Solar Array Electrical Power Curve

Figure 3.3-5 shows plots for power and current vs. voltage for the baseline solar array design. The plot illustrates the ideal operating point of the system occurs near 28V, where power generated is at a maximum.



Source: MDL January 2008

Figure 3.3-5 Solar Array Current and Power Curves

5.3.2.1.4 SA Magnetic Design

The Solar Array can be a very significant source of stray magnetic fields in the spacecraft due to the large currents[9]. On the other hand, its linear geometry makes it the most straightforward one to compensate or cancel out by correct placement of forward and return interconnections. The stray fields can be minimized by using a technique referred to as "backwiring". In backwiring the return wire from each string of solar cell modules is returned directly underneath the modules in that particular string and carefully routed along a line just behind the centerline of the modules. Each string and module of the string is self-canceling and does not depend on the magnetic field of an adjacent module or string for cancellation. If a module fails in flight the current in both the string and the return drop to zero simultaneously leaving no uncompensated currents in the array. Some contemporary arrays use string-switching techniques that effectively change the current path dynamically in response to load and solar input changes. Unless backwiring is used it is extremely difficult to totally compensate for the solar array magnetic field[10].

3.3.2.2 BATTERY

Solar array power will be supplemented by a low Ah battery during LEOP, anomalous array eclipses, and during intermittent peak power consumption periods expected to occur during science observations. A 20 Ah Li-Ion battery is the current baseline battery based on estimated LEOP and cruise power requirements, estimated sun acquisition time, and depth of discharge. The very low usage that the battery will receive will allow a depth of discharge of roughly 42% to be tolerated. Li-Ion batteries are easily capable of surviving the 18 month cruise phase. Battery sizing parameters and calculations are provided in Table 3-9.

Table 3-96: Battery Sizing Calculations

Battery Sizing Calculation Parameters		
Parameter	**Value**	**Comment**
Cruise power requirement	383.6 W	
Cruise Depth of Discharge	42%	
Cruise Sun acquisition duration	37 min (.617h)	
Bus Voltage	28 V	
Launch Depth of Discharge	32.8%	
Launch Energy Storage Req't (W-hr)	183.5 W-hr	
Launch Sun acquisition time	60 min	
Battery Size Calculations		
Cruise Phase Analysis		
Parameter	**Equation**	**Value**
Cruise Energy (W-hr)	383.6W * .617	236.7Wh
Cruise amp-hours	236.7Wh / 28V	8.45 Ah
Dod Corrected Battery Size	8.45 A-hr / .42	20 A-hr
Launch Analysis		
Parameter	**Equation**	**Value**
Launch amp-hours	183.5 W-hr/28V	6.55 A-hr
DoD Corrected Battery Size	6.55 A-hr/.328	20 A-hr

A Li-Ion battery similar to the one used on NASA's Solar Dynamics Observatory (SDO) shown in Figure 3.3-6 would be used in the Sciencecraft EPS.

Figure 3.3-6: Li-Ion Battery

3.3.2.3 Power Control and Distribution System

A traditional S3R power control design uses switching to shunt sections of the array to control power. Electrical fluctuations and thermal disturbances caused by shunting array sections make the S3R power control system incompatible with the LISA payload science requirements. A maximum peak power tracker (MPPT) system is the current baseline power control system due to its ability to adjust the operating point of the array without switching and thus ensure thermal and electrical stability. Another advantage to the MPPT design would be a much simpler stringing design due to a simpler thermal environment.

Details of the baseline MPPT power control design can be found in the 2000 LISA Final Technical Report[11].

3.4 ATTITUDE CONTROL SYSTEM (ACS) DESIGN

Mission baseline parameters driving the Sciencecraft Attitude Control System design are referenced in Table 3-11.

Table 3-7: Sciencecraft ACS Baseline Parameters

Item/Function	Requirement	Comments
Total Micropropulsion Impulse	8300 Ns per thruster	Total propellant required based on 8.5 years of operation (micropropulsion system is not used during transfer)
Attitude sensing	1 arcsec RMS 3σ TBR	Ensure sufficient sensing accuracy during laser beam acquisition phase
Contamination	The payload must be protected from contamination at all times	The thruster plumes from all elements of the ACS system must not impinge on or contribute to contamination of the payload

3.4.1 ACS OVERVIEW

Sciencecraft ACS responsibility begins after separation from the P/M. Prior to this separation, the Sciencecraft has been placed in the science mode orbit with a proper alignment to the sun and has been spin stabilized to provide attitude stability with respect to the sun line. It is the responsibility of the Sciencecraft ACS to gradually null the S/C rates and to maintain attitude stability up until commencement of laser acquisition procedures. During the Sciencecraft commissioning phase, the Sciencecraft will use either Coarse Sun Sensors (CSS) or Star Trackers (ST) for relative/absolute attitude sensing, and rate gyros to sense attitude rates. Once the process of establishing the optical links between the three Sciencecraft commences, attitude control will fall within the domain of the Disturbance Reduction System (DRS). The DRS is responsible for drag-free control of the LISA constellation during science observations. Should an anomalous event cause a loss of laser phase-lock, the Sciencecraft would once again rely upon the bus ACS to restore the system to a safe-hold mode until laser phase-lock can be re-established.

3.4.2 ATTITUDE CONTROL SYSTEM ARCHITECTURE

3.4.2.1 COARSE SUN SENSORS

Preferred over digital sun sensors for their lower mass and cost, coarse sun sensors will be more than adequate for safe-holding the Sciencecraft while the STs provide the first line of defense for anomaly detection. Twelve coarse sun sensors will be positioned around the Sciencecraft, configured into two redundant strings of six sensors. Six additional sensors will be located on the P/M. Both Sciencecraft strings will have sensors located around the circumference of the Sciencecraft bus to provide full sky coverage. The use of coarse sun sensors will require an analog interface card to be provided in the C&DH unit. Figure 3.4-2 illustrates the mounting locations for the Sciencecraft coarse sun sensors.

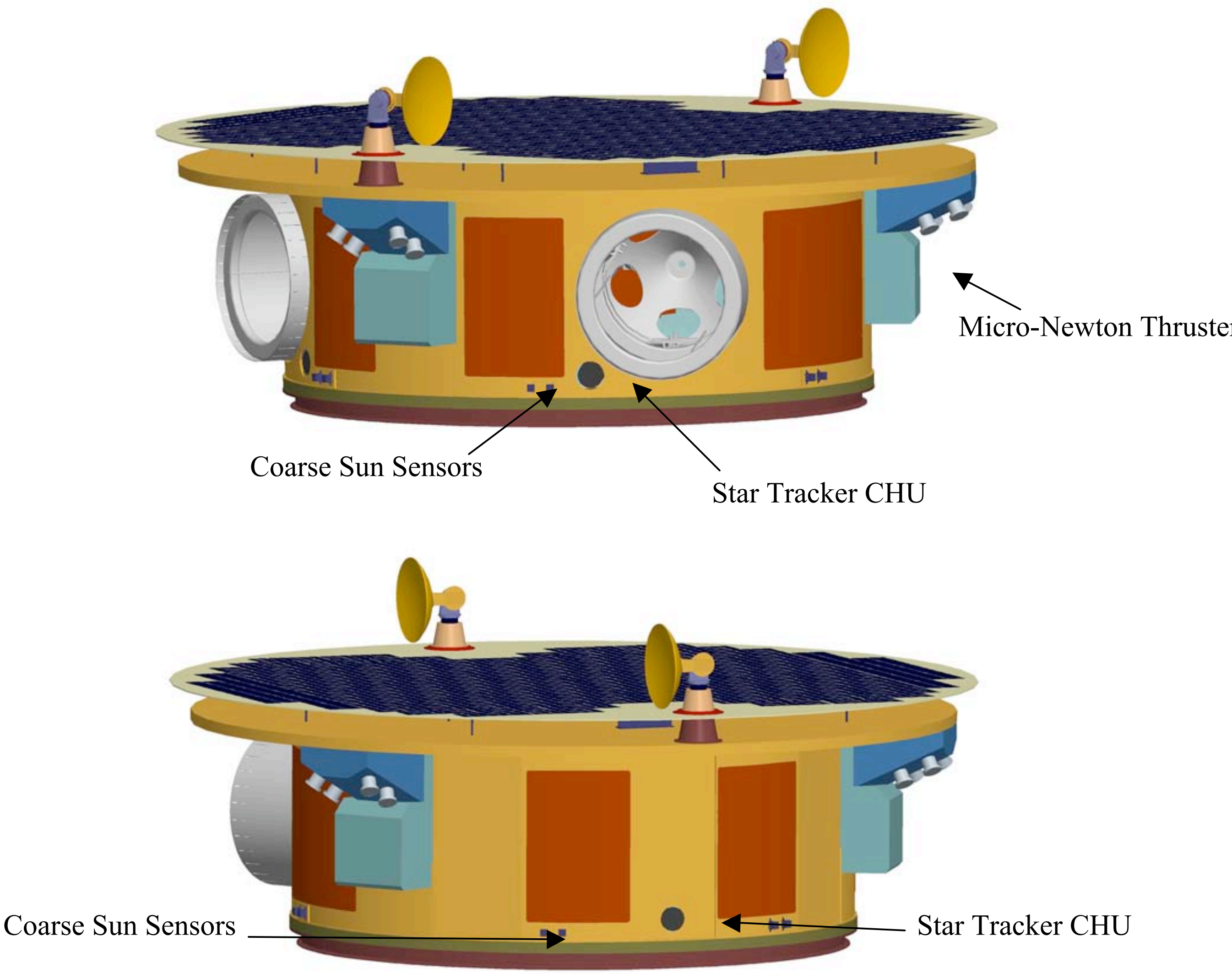


Figure 3.4-2: CSS, Star Tracker CHU and Micro-Newton Thruster Mounting Locations

3.4.2.2 Star Trackers

The ACS will use a Star Tracker system consisting of two ST processing units and five ST camera head units (CHUs). One ST processing unit is required to operate the system while the second unit will provide redundancy. Two ST CHUs will be located near the telescope assemblies on the Sciencecraft and are required for system functionality. The third ST CHU is provided for redundancy and will be located on the rear exterior wall of the Sciencecraft bus. With the P/M outer shell blocking Sciencecraft ST CHU visibility, two additional ST CHUs will be mounted on the P/M to provide visibility during the cruise phase. Cross-strapping between the ST processing units and head units will provide additional reliability to the system. Figure 3.4-2 shows ST CHU mounting locations on the Sciencecraft. The ST processing units will be mounted on the bottom deck of the Sciencecraft.

3.4.2.3 Rate Sensing Gyros

Rate gyros are required to provide attitude rate data during P/M separation and subsequent rate nulling, as well as to provide attitude rates during Level-II (hardware) safe-hold mode. Rate information will also be provided by the STs. ST derived rates will be used during Level-I (Software) safing, as well as other modes. However, the need for a direct 3-axis rate sensing ability that can be relied upon throughout the mission is partially driven by the level of confidence that can be placed on the rate information derived from the STs in the event of an apparent dynamic anomaly during an ST-based control mode. Two sets of gyros will be used in hot-redundancy to provide attitude rate measurements. Solid state gyros with no moving parts are required for their reliability and minimum disturbances they impart upon the system. The gyro units will be mounted on the bottom deck of the Sciencecraft as shown in Figure 3.4-3.

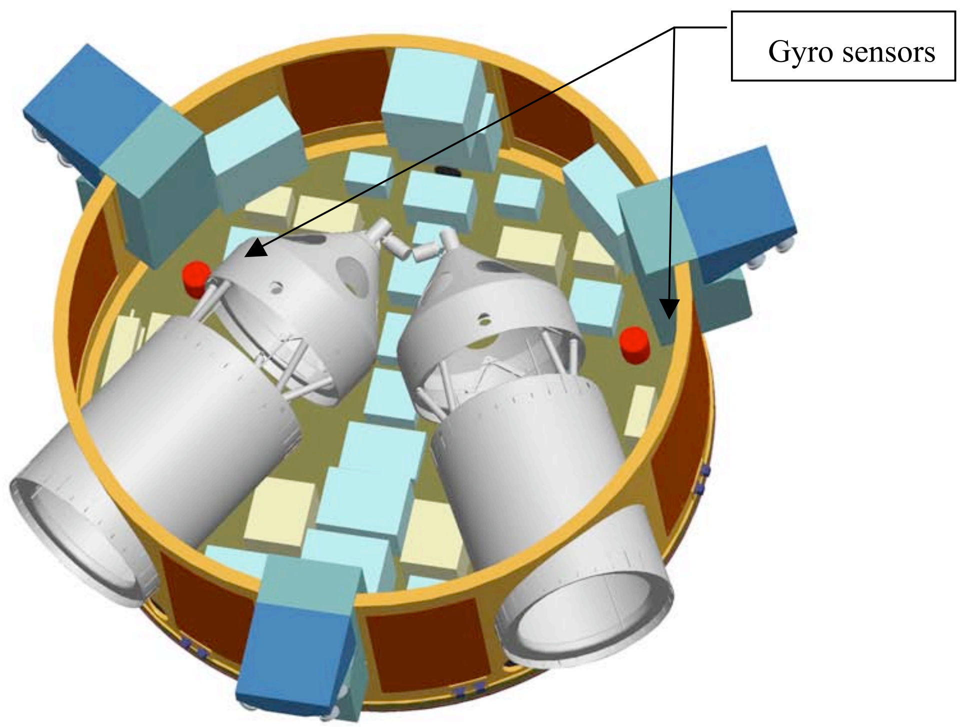


Figure 3.4-3: Rate Gyro Mounting Locations

3.4.2.4 ACS Actuators

The Micro-Newton thrusters used by the DRS during science observation will also function as the ACS actuator system on the Sciencecraft. The Micro-Newton thrusters are not used for constellation station keeping, as the orbits are optimized to maintain acceptable constellation formation during the coarse of the LISA mission.

The Micro-Newton thrusters are configured as clusters of emitters, each arranged on the faces of a pyramid. To minimize the potential for contamination of the telescopes by the thruster plumes, the location and orientation of the thruster emitters are optimized to maximize the separation between the thrust axis and the telescope FOVs. Refer to the Sciencecraft Mechanical Systems section of this document for details about Micro-Newton thruster mounting and plume analysis.

3.4.3 Spin Stabilization

One crucial event during the LISA mission will be the Sciencecraft deployment procedure. Having only the Micro-Newton thrusters to correct attitude during separation from the P/M, post-separation tumble would be inevitable. Designing for post-separation tumble would require increasing battery size to compensate for solar array eclipsing and would require additional hardware to protect sensitive optical components from sun exposure. Spin stabilization will ensure that post separation tumble does not occur by holding the excursions of the Sciencecraft arrays with respect to the sun line to an acceptable range. Current analysis, based on estimated S/C inertia and typical separation tip-off rates, indicates that a 2.1 deg/sec spin rate would provide the necessary inertial stability to ensure safe separation and spin down. ACS thrusters on the P/M will be used to generate the spin. The Micro-Newton thrusters will be used to de-spin the Sciencecraft. Current analysis based on worst case conditions and significant margins on Sciencecraft inertia and tip off rates indicates a maximum de-spin duration of 5 days. Although the de-spin period will delay commencement of science observations, the required Micro-Newton thruster impulse would not otherwise impact or degrade the mission.

3.4.4 LISA Mission ACS Phases and Modes

The LISA Mission is divided into four distinct ACS phases:

1. **LEOP & Cruise Phase** - 3-axis controlled LCM employing P/M thrusters
2. **Orbit Insertion & P/M Separation Phase** - Use P/M thrusters for final burn to insert the Sciencecraft into the proper orbit. Use P/M thrusters to spin up the Sciencecraft along the Sun Line and separate from the P/M.
3. **Constellation Commissioning Phase** - Commission the Micro-Newton thrusters. Null out rotational rates of the Sciencecraft and commence commissioning of payload and various subsystems.
4. **Science Operations Phase** - Commence signal acquisition procedure to phase lock the three arms of the LISA constellation. The payload inertial sensor is fully active and the DRS controls are active.

These three phases are further divided into 11 distinct ACS modes:

1. Sun Acquisition Mode (SAM)
2. Transfer Orbit Safe Mode (TOSM)
3. Star Sensor Pointing Mode (SSPM)
4. Cruise Mode (CM)

5. Operational Attitude Acquisition Mode (OAAM)
6. Star Sensor Working Mode (SSWM)
7. Operational Orbit Safe Mode (OOSM)
8. Drag Free Acquisition Mode (DFAM)
9. Laser Beam Acquisition Mode (LBAM)
10. Instrument Commissioning & Calibration Mode (ICCM)
11. Science Mode (SM).

The LISA mission ACS modes and inter-relationships between the modes are shown in Figure 3.4-4.

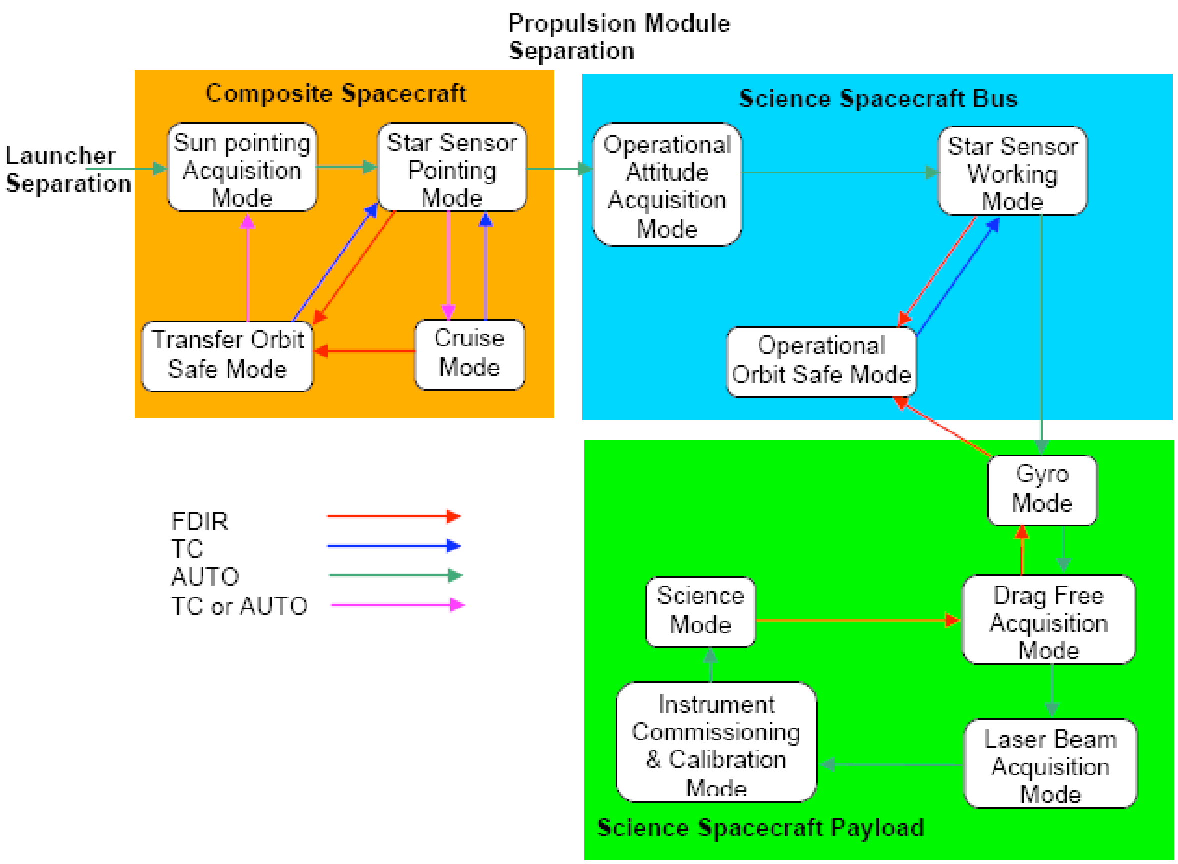


Figure 3.4-4: ACS Modes

3.4.5 DISTURBANCE REDUCTION SYSTEM

The disturbance reduction system is responsible for the dynamic control of all spacecraft and proof masses within the LISA constellation, from initial laser signal acquisition and throughout the science observations. It consists of five control functions:

1) Attitude control system (ACS): to orient the S/C to align the telescopes with incoming laser beams

2) Drag free control system (DFC): to maintain drag free motion of the proof masses in LISA measurement directions

3) Proof mass suspension control: to maintain relative attitude of the proof mass with respect to its housing and to maintain relative position of the proof mass with respect to its housing in the transverse directions

4) Telescope articulation control: to maintain the angle between the telescopes

5) Point-ahead (PA) control and acquisition: to point the outgoing beam while sensing the incoming beam

3.4.5.1 ACQUISITION

Initial acquisition of laser signals within the LISA constellation will commence once the Sciencecraft bus ACS system has positioned and oriented all three Sciencecraft within close alignment for optical linking to occur. Lasers signals will then be sent from each Sciencecraft payload in a specific sequence. Software algorithms will facilitate controlled sweeps of each outgoing laser until the signal is detected by the interferometer on the receiving end of the link. The acquisition phase will be complete once all six interferometer links are established.

3.4.5.2 CONTROL DURING SCIENCE MODE

The basic approach is to use the inertial wavefront sensing provided by the science interferometer for Sciencecraft attitude control. Using proof mass metrology, this approach utilizes optical sensing in 3 degrees of freedom and capacitive sensing in the remaining three to control the relative position of the Sciencecraft and proof masses for drag-free flight within the LISA Band. All degrees of freedom of the proof masses, except for the LISA sensitive axes, are controlled by the electrostatic suspension actuation below the LISA band. Micro-Newton thrusters are used to control the position and attitude of the Sciencecraft. An articulation mechanism is used to provide in-plane rotation of the articulating telescope. The complete list of sensors and actuators used in the DRS are provided in Table 3-12.

Table 3-8: Sensors and Actuators in the DRS

Sensors	Actuators
IWS	Micro Newton Thrusters
Capacitive sensing	Electrostatic Suspension
Optical sensing	Articulation Mechanism
Star Tracker (Acquisition Only)	Point-Ahead Mirror Mechanism
Articulation Encoder (Acquisition Only)	

3.4.5.3 SUSPENSION CONTROL

There are several modes of operation for GRS suspension. In the science mode, all proof-mass degrees of freedom, except for the one along the LISA sensitive axis, are stabilized by using electrostatic suspension control of the GRS. This is especially necessary due to the inherent electrostatic-induced stiffness of the proof-mass dynamics, which may otherwise yield an unstable system. Note that during the laser acquisition process the suspension control is in the so-called "accelerometer mode", wherein the control loop is considerably stiffer along the rotational axes such that the GRS can serve as an angular accelerometer. The accelerometer is used in concert with the high-resolution star tracker to stabilize the spacecraft during acquisition's blind mode.

3.4.5.4 DRAG FREE CONTROL

Along the two sensitive axes, the Sciencecraft is controlled around the proof-masses such that all residual accelerations in the sensitive axes are minimized. In addition, the Sciencecraft follows the average out-of-plane motion of the proof-masses in order to compensate the solar dynamic pressure. The control error signals are obtained from the optical and electrostatic readout system of the GRS. The optical proof-mass metrology is used on the two sensitive axes in order to reduce the sensor noise in the generated force signals to the Micro-Newton propulsion system. The drag-free control covers 3 degrees of freedom of the Sciencecraft.

3.4.5.5 SCIENCECRAFT ATTITUDE CONTROL

During science mode, the Sciencecraft attitude control is performed by feeding back the information from inertial wavefront sensing (IWS) to the Micro-Newton propulsion system. Since two telescopes are on-board and mounted at an angle of nominally 60 deg with respect to each other, IWS provides a total of 4 tip and tilt error angles. By applying the corresponding geometric relations, these angles can be used to determine the complete Sciencecraft attitude error to align the telescopes with respect to the incoming laser beams as well as the angular error between the two telescopes. The former is used in the ACS loop to properly point the Sciencecraft. This use of IWS is a deviation from the LISA Pathfinder concept made possible by the LISA constellation configuration. IWS is the most accurate attitude information method that benefits by having a very low noise level.

3.4.5.6 TELESCOPE ARTICULATION CONTROL

Due to orbital mechanics, the angle between the two interferometer arms is constantly changing. Therefore, this angle must be controlled as well. The information from the IWS is used again for this purpose. The 4 tip and tilt angle errors from IWS can be used to determine the complete Sciencecraft attitude error with respect to the incoming laser beams as well as the angular error between the two telescopes. The latter one is used to generate a feedback signal for the telescope actuators. Note that one of the two telescopes will remain in a fixed position (a cold spare) while the second one will be constantly actuated at 10 Hz.

3.4.5.7 POINT AHEAD ACTUATOR (PAA) CONTROL

The LISA formation is not a stationary one. In fact, because of the natural orbits of the three Sciencecraft, the formation plane breaths and tilts while the formation angles oscillate at orbital rates. This means that each Sciencecraft is moving relative to the other two. Given that the Sciencecraft are roughly 5 million km apart from each other, and that the power of the 1W laser is reduced to around 100 pW at that distance, it is imperative that each telescope points to where the other Sciencecraft would be in the time it takes for the light to go from one Sciencecraft to the other. In other words, each telescope must point ahead to where the other Sciencecraft will be. This is accomplished by a point-ahead mirror that is actuated by a piezo-based drive. The mirror is actuated out-of-plane only, since that component of the point-ahead angle exhibits large variations throughout the year. The in-plane component shows a small variation about a fixed bias, which may be accommodated by a pre-fixed tilt of the mirror.

3.4.6 ACS DESIGN ALTERNATIVES

3.4.6.1 STAR TRACKER CAMERA HEAD REDUCTION OPTION

The baseline ST configuration requires two additional ST CHUs mounted on the P/M due to the FOV obstruction caused by the P/M shell during the cruise phase. An alternative design would have viewing ports added to the P/M shell to allow the Sciencecraft ST CHUs to function while nested in the P/M. This would allow elimination of the P/M mounted ST CHUs which could yield both weight and cost benefits and a risk reduction. This design would require intentional misalignment between the ST head units and the telescopes to prevent damage to sensitive optical equipment from sun exposure during the cruise phase. Analysis of the ST viewing port impact on the P/M structural integrity will have to be conducted.

3.4.6.2 MEMS GYRO OPTION

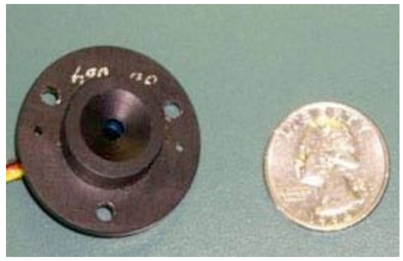
The use of Micro Electro-Mechanical Sensor (MEMS) gyros is currently being considered as an alternative design that would reduce gyro system weight, power requirement and possibly cost. The coarse rate measurement offered by the MEMS gyro design would suffice to reduce the rate to a range in which the ST system can take over, but may not be acceptable for long-term standalone rate sensing. Limited flight heritage and possible NRE costs of the MEMS gyro design will have to be considered.

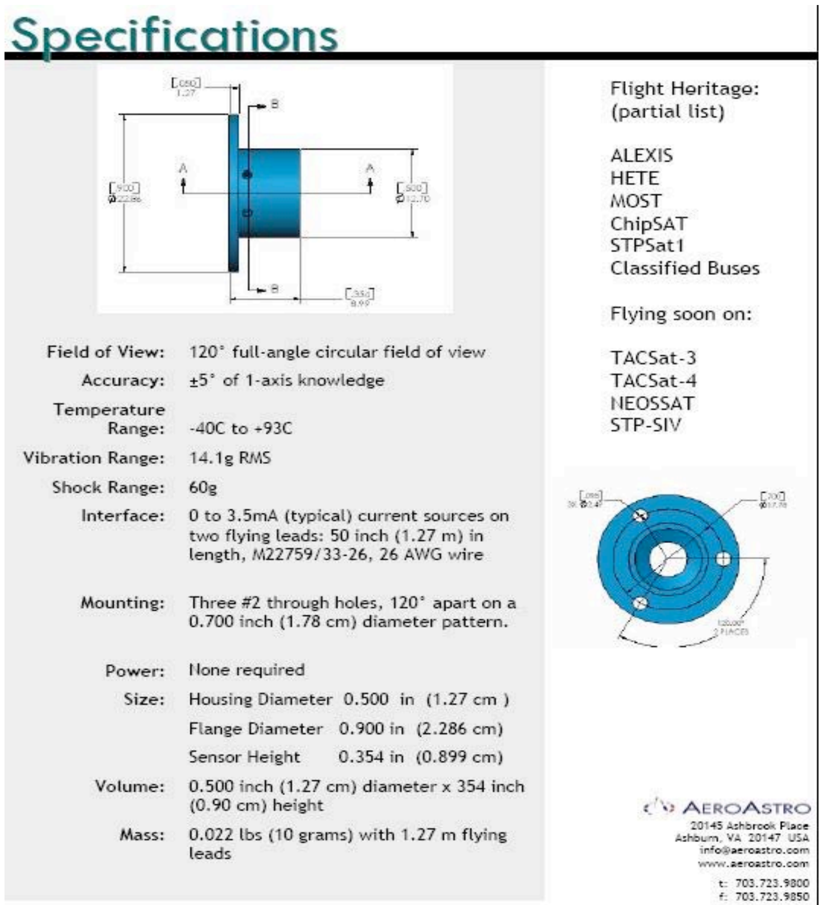
3.4.7 ACS HARDWARE

The components and performance specs listed below represent candidate hardware that might be used in the final LISA ACS design. Hardware information accuracy will improve as the ACS design matures.
Assume one operating processing unit and two operating CHUs.

3.4.7.1 COARSE SUN SENSORS

A candidate supplier for the coarse sun sensors is AeroAstro. Dimensional information and specifications for the candidate CSS is provided in Figure 3.4-5





Specifications

Flight Heritage: (partial list)

ALEXIS
HETE
MOST
ChipSAT
STPSat1
Classified Buses

Flying soon on:

TACSat-3
TACSat-4
NEOSSAT
STP-SIV

Field of View:	120° full-angle circular field of view
Accuracy:	±5° of 1-axis knowledge
Temperature Range:	-40C to +93C
Vibration Range:	14.1g RMS
Shock Range:	60g
Interface:	0 to 3.5mA (typical) current sources on two flying leads: 50 inch (1.27 m) in length, M22759/33-26, 26 AWG wire
Mounting:	Three #2 through holes, 120° apart on a 0.700 inch (1.78 cm) diameter pattern.
Power:	None required
Size:	Housing Diameter 0.500 in (1.27 cm)
	Flange Diameter 0.900 in (2.286 cm)
	Sensor Height 0.354 in (0.899 cm)
Volume:	0.500 inch (1.27 cm) diameter x 354 inch (0.90 cm) height
Mass:	0.022 lbs (10 grams) with 1.27 m flying leads

AeroAstro
20145 Ashbrook Place
Ashburn, VA 20147 USA
info@aeroastro.com
www.aeroastro.com
t: 703.723.9800
f: 703.723.9850

Figure 3.4-5: Coarse Sun Sensor

3.4.7.2 STAR TRACKERS

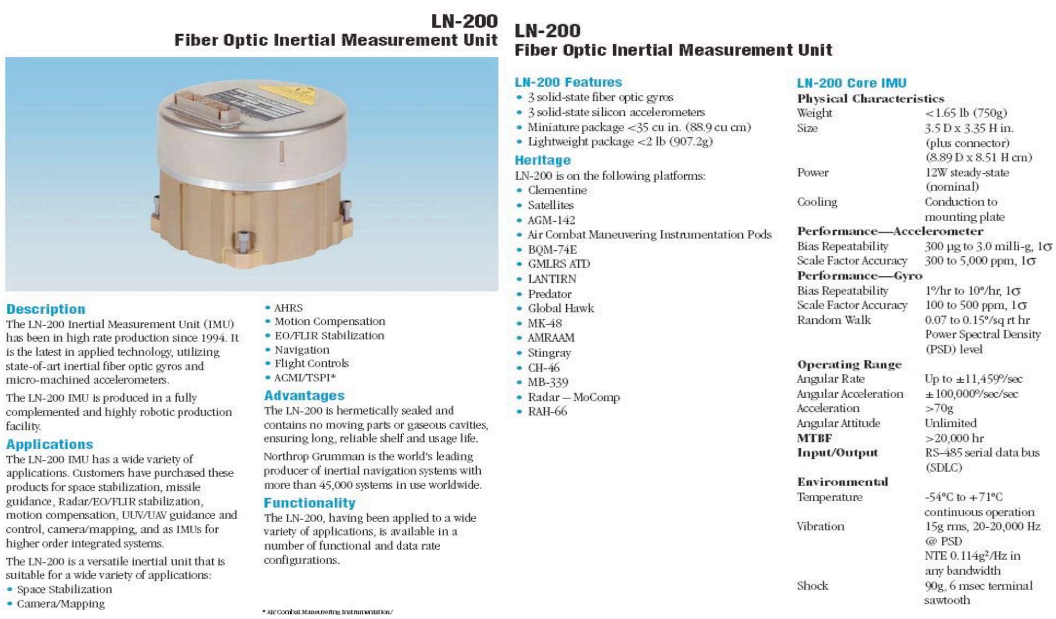
A candidate supplier for the ST system would be Denmark Technical University (DTU). A Star Tracker System consisting of the electronics unit, CHU and light baffle similar to what might be used in the final design are shown in Figure 3.4-6.

Specs:
Accuracy: 1 arcsec at 3σ
Tracking Rate: 8 deg/sec
FoV: 18.4 deg x 13.4 deg
Refresh Rate: 10 Hz
Data Interface: RS-422
Mass: 1.1 kg
Power: 3.8W - Processor
.7W - CHU
TRL = 9

Figure 3.4-6: Star Tracker

3.4.7.3 GYRO

The Northrup Grumman Inertial Measurement Unit (IMU) model LN-200 is the selected rate gyro for the LISA baseline design. An illustration of the baseline gyro with performance parameters is provided in Figure 3.4-7.



LN-200
Fiber Optic Inertial Measurement Unit

Description

The LN-200 Inertial Measurement Unit (IMU) has been in high rate production since 1994. It is the latest in applied technology, utilizing state-of-art inertial fiber optic gyros and micro-machined accelerometers.

The LN-200 IMU is produced in a fully complemented and highly robotic production facility.

Applications

The LN-200 IMU has a wide variety of applications. Customers have purchased these products for space stabilization, missile guidance, Radar/EO/FLIR stabilization, motion compensation, UUV/UAV guidance and control, camera/mapping, and as IMUs for higher order integrated systems.

The LN-200 is a versatile inertial unit that is suitable for a wide variety of applications:

- Space Stabilization
- Camera/Mapping
- AHRS
- Motion Compensation
- EO/FLIR Stabilization
- Navigation
- Flight Controls
- ACMI/TSPI*

Advantages

The LN-200 is hermetically sealed and contains no moving parts or gaseous cavities, ensuring long, reliable shelf and usage life.

Northrop Grumman is the world's leading producer of inertial navigation systems with more than 45,000 systems in use worldwide.

Functionality

The LN-200, having been applied to a wide variety of applications, is available in a number of functional and data rate configurations.

* Air Combat Maneuvering Instrumentation/

LN-200
Fiber Optic Inertial Measurement Unit

LN-200 Features

- 3 solid-state fiber optic gyros
- 3 solid-state silicon accelerometers
- Miniature package <35 cu in. (88.9 cu cm)
- Lightweight package <2 lb (907.2g)

Heritage

LN-200 is on the following platforms:

- Clementine
- Satellites
- AGM-142
- Air Combat Maneuvering Instrumentation Pods
- BQM-74E
- GMLRS ATD
- LANTIRN
- Predator
- Global Hawk
- MK-48
- AMRAAM
- Stingray
- CH-46
- MB-339
- Radar – MoComp
- RAH-66

LN-200 Core IMU

Physical Characteristics	
Weight	<1.65 lb (750g)
Size	3.5 D x 3.35 H in. (plus connector) (8.89 D x 8.51 H cm)
Power	12W steady-state (nominal)
Cooling	Conduction to mounting plate
Performance—Accelerometer	
Bias Repeatability	300 μg to 3.0 milli-g, 1σ
Scale Factor Accuracy	300 to 5,000 ppm, 1σ
Performance—Gyro	
Bias Repeatability	1°/hr to 10°/hr, 1σ
Scale Factor Accuracy	100 to 500 ppm, 1σ
Random Walk	0.07 to 0.15°/sq rt hr Power Spectral Density (PSD) level
Operating Range	
Angular Rate	Up to ±11,459°/sec
Angular Acceleration	±100,000°/sec/sec
Acceleration	>70g
Angular Attitude	Unlimited
MTBF	>20,000 hr
Input/Output	RS-485 serial data bus (SDLC)
Environmental	
Temperature	-54°C to +71°C continuous operation
Vibration	15g rms, 20-20,000 Hz @ PSD NTE 0.114g²/Hz in any bandwidth
Shock	90g, 6 msec terminal sawtooth

Figure 3.4-7: Gyro

3.5 PROPULSION (ON-BOARD) DESIGN

LISA requires micro Newton thrusters to provide the fine spacecraft attitude and position control for drag free flight and beam pointing to the distant spacecrafts. The thrusters are operated continuously during science operations with their thrust levels set by the disturbance reduction system control loops. Three different thruster technologies are nearing flight readiness and are currently capable of meeting the LISA thrust and thrust noise requirements: the colloid micro Newton thruster (CMNT) made by Busek Co. in Boston, the indium needle field emission electric propulsion (In-FEEP) thruster made by ARC Seibersdorf in Austria, and cesium slit FEEP (Cs-FEEP) made by ALTA S.p.A. in Italy. These are a part of the LISA technology development effort and are described in detail in the "LISA Technology Status Report"[12].

At least six micro-thrusters on each spacecraft must be operating continually during science operations for the entire LISA mission. Enough consumables must be carried for the entire extended mission. With sunlight photon pressure as the largest disturbance acting on the spacecraft, the micro-thrusters must produce on the order of 10 µN of thrust with better than a 0.1 µN resolution during science measurements. Furthermore, over the LISA science measurement bandwidth, thrust and thrust noise must be stable and within the error limitations of the DRS over the entire mission, < 0.1 µN/√Hz (open loop) at the high end of the measurement band. Brief periods of higher thrust, >30 µ N, may be required during tip-off recovery, constellation acquisition, and safe-mode operations; however, lifetime and thrust noise requirements do not apply to these conditions. Finally, the micro-thrusters cannot create harmful interactions with the spacecraft such as charging or contamination of spacecraft surfaces. The micro-thruster requirements are summarized in Table 3-13.

Table 3-9: Key Micro-Newton Thruster Performance Requirements

Parameter	Req't	Units	Comments
Science Mode Thrust Range	Min: 4 Max: 20	µN	Both Colloid and FEEP thruster emitters ca to meet any thrust range requirement
Safe Mode/Tip Off Thrust Range	Min: 4 Max: > 30	µN	
Science Mode Thrust Precision	0.1	µN	Thrust precision is dictated by the bit-resol thruster electronics
Science Mode Open Loop Thrust Nois	0.1	µN/√Hz	Relaxation of this requirement is possible a frequencies (TBR)
Operational Lifetime	44,000	hours	
Total Impulse per Thruster	3,000	Ns	
Contamination	0.1	$\mu g/cm^2$	

The most advanced thruster technology that can meet these requirements falls into the category of field emission or *electrospray propulsion*, shown schematically in Figure 3.5-2. In these thrusters, the propellant is a conductive liquid fed to a sharpened emitter where a balance between high electric fields and surface tension forces produce ions and/or charged droplets. The charged particles are subsequently accelerated to high velocity (1-100 km/s) by the same or an additional electric field, and are neutralized downstream by the emission of electrons from a separate cathode. The working dimensions of each electrospray emission point are on the order of microns, with emission current levels between 0.2 – 100 µA. Emission points can be combined together along a slit geometry between two plates (Cs-FEEP) or in multiple single-emitters fed through capillary tubes (CMNT) or along the outside surface of roughened needles (In-FEEP). Thrust is determined by controlling the total beam voltage, V_b, and current, I_b, (related to mass flow rate, $I_b = \dot{m}(q/m)$),

$$Thrust \propto I_b \sqrt{V_b \left(\frac{m}{q} \right)}$$

where m and q are the mass and charge of a single ion or droplet. For thrusters that emit charged droplets (usually near the Rayleigh break-up limit), the mass-to-charge ratio is proportional to the current and dependant on the physical parameters of the propellant (density, conductivity, viscosity, surface tension, etc.). Propellant flow rate is either controlled actively by the use of a precision regulating valve (CMNT) or passively by capillary action (both FEEP thrusters). For thrusters with passive propellant feed systems, the flow rate and current is also dependant on the voltage and physical properties of the propellant. In all thruster configurations, the thrust resolution and noise is simply governed by how precisely current and voltage can be controlled while the physical properties of the propellant are kept constant, usually requiring active temperature control.

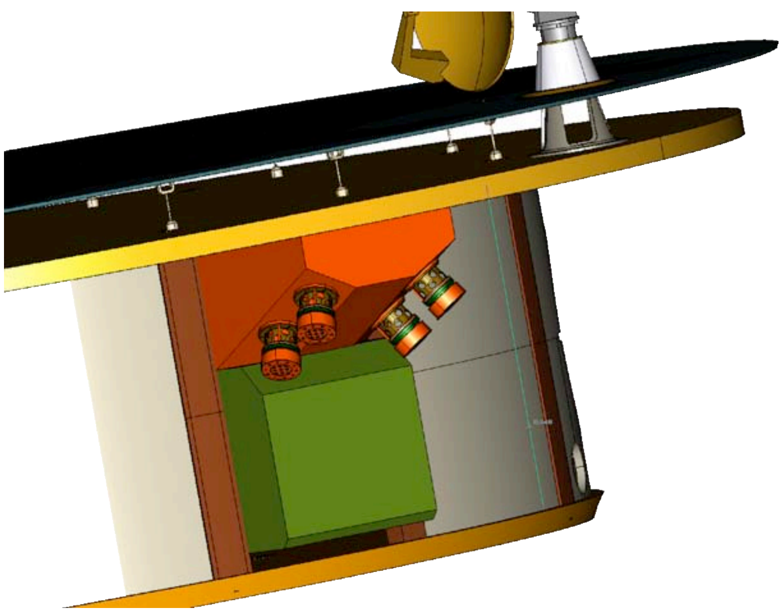
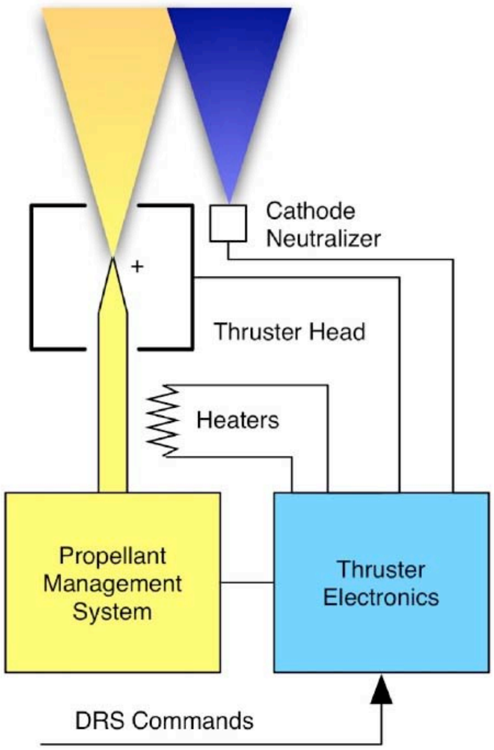


Figure 3.5-1: Micro-Colloidal Thruster-Cluster Location



The most advanced thruster technology that can meet LISA requirements falls into the category of field emission or electrospray propulsion. Pictured here is a typical block diagram of electrospray thruster system.

Figure 3.5-2: Electrospray Thruster System Block Diagram

The NASA component of the micro-thruster technology effort focuses on the colloid micro Newton thruster (CMNT), currently being developed and qualified by Busek Co. and JPL for the Space Technology 7 Disturbance Reduction System (ST7-DRS) mission that will be part of LISA Pathfinder. The CMNT uses a capillary emitter fed by a piezoelectric micro valve with an ionic liquid as the propellant (a single emitter is shown in Figure 3.5.3).

3.5.1 Colloid Micro-Newton Thruster (CMNT)

NASA is developing the colloid micro Newton thruster (CMNT), led by JPL and Busek Co. and for the Space Technology 7 Disturbance Reduction System (ST7-DRS) mission that will be part of LISA Pathfinder. The CMNT uses a capillary emitter fed by a piezoelectric microvalve with an ionic liquid as the propellant (an operating single emitter is shown in Figure 3.5.3, along with a simplified schematic).

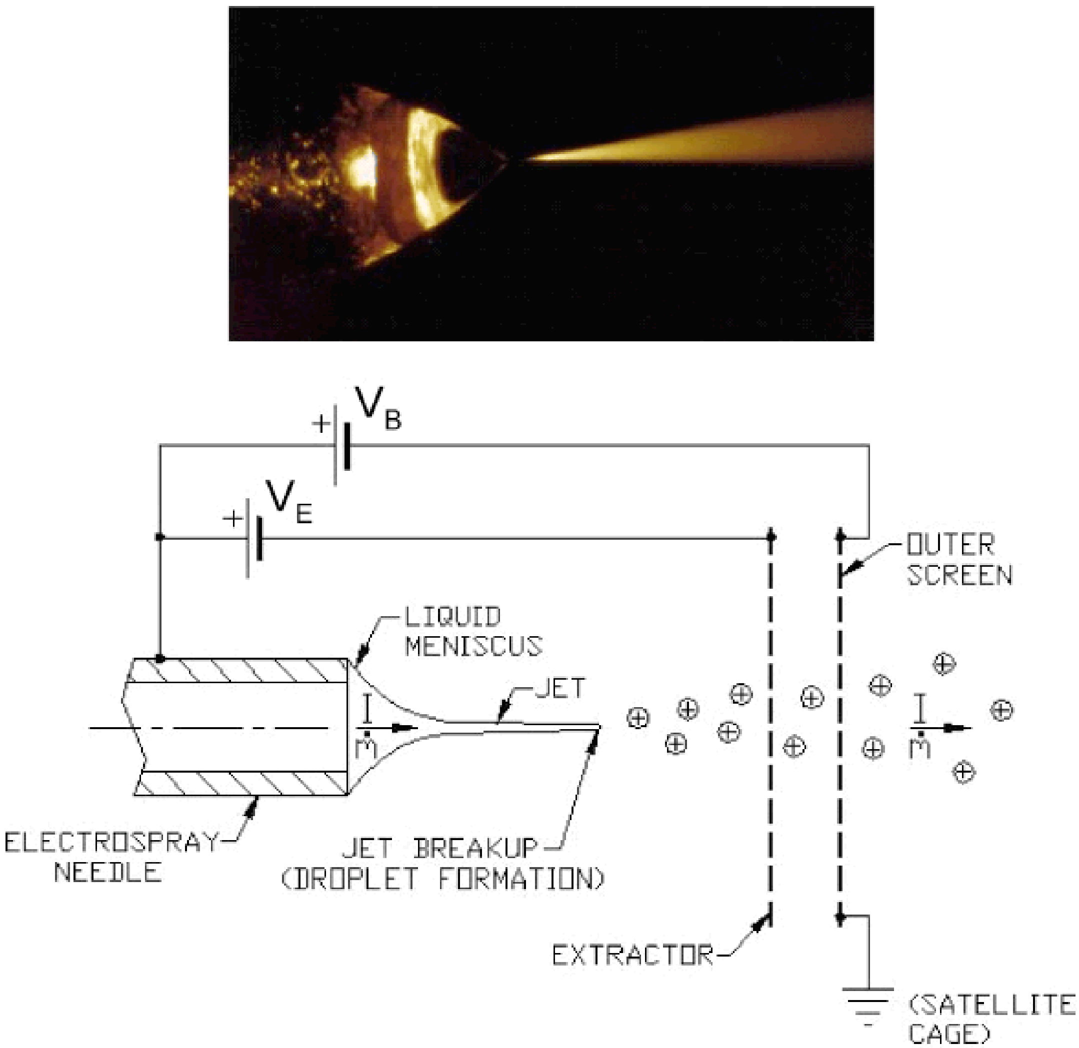


Figure 3.5.3: Colloid Thruster Single Emitter and Simplified Schematic

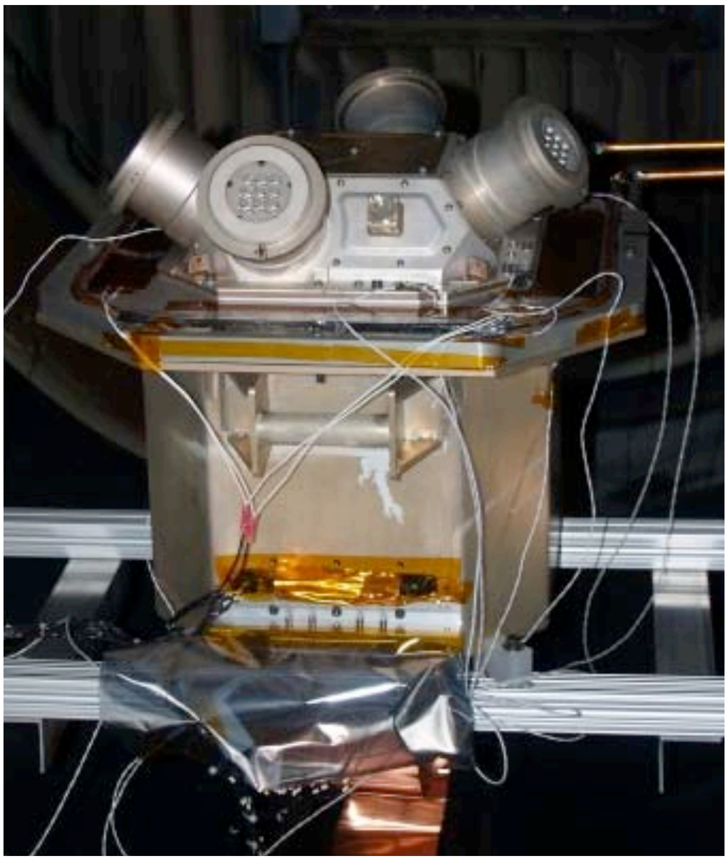


Figure 3.5-4: ST7 CMNT Cluster Flight Hardware in Thermal Vacuum Qualification Test

The CMNT is in the final stages of flight qualification for ST7-DRS, with a flight cluster shown in Figure 3.5-. Flight-like engineering model (EM) versions of the CMNT have already demonstrated the ST7 and LISA performance requirements through direct measurement of thrust and calculation of thrust noise. Figure 3.5-4 shows a plot of thrust noise, based on recent beam current and voltage measurements, of an EM unit in current control mode.

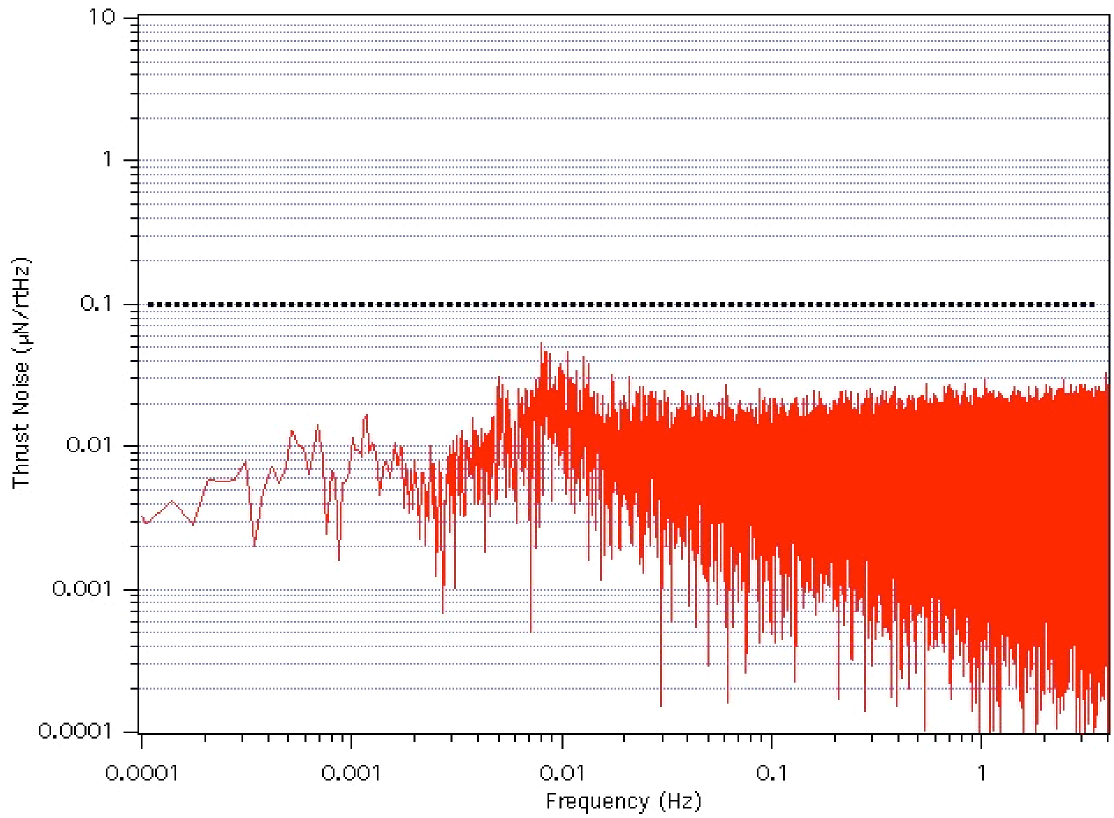


Figure 3.5-4: ST7 CMNT Thrust Noise Plot

Models and measurements of breadboard and EM-level CMNT exhaust plumes have shown that no charged particles exit the thruster beyond a 35-degree half-angle at the most divergent operating condition. Mass deposition measurements have shown that no measurable deposition occurs outside of a 45 degree half-angle, with more detailed measurements at various angles using flight-like EM units to be completed by the end of 2006. Plasma potential measurements of a single-emitter system show that beam potentials do not exceed +50 V at any angle without using a neutralizer. To date, all tests show that the Busek CMNT will meet LISA contamination requirements. Still, as the diagnostics are already developed, measurements will continue during the long duration lifetime demonstration tests to insure the plume characteristics (which can effect both performance and contamination) do not change over the thruster lifetime. Five 3000-hour class tests of breadboard and EM-level thruster units have been completed successfully at Busek for ST7-DRS, which has a lifetime requirement of 2200 hours. The first series identified and corrected a needle wear mechanism associated with propellant electrochemistry and the needle material. The next two tests identified and corrected a new problem of gradual clogging near the emitter tip by changing the emitter geometry and material while improving emitter manufacturing processes. The fourth test demonstrated a slight modification to emitter geometry that improved thrust stability and susceptibility to bubbles in the feed system. The final test included a full EM-level system with a flight-like thruster head, feed system, and electronics. This test showed that longer durations than the ST7-DRS lifetime can be supported with the unit reaching >3400 hours of continuous operation as of November 2006, without failure

The baseline LISA CMNT architecture includes two working thruster systems and two redundant systems, as shown in Figure 3.5-5. All the components, including the electronics, are based on the proven ST7 architecture and designs for the four-thruster cluster. New spherical metal diaphragm tanks will reduce the mass of cluster significantly, providing enough propellant for the entire plus extended mission for two thrusters firing continuously. Any other component requiring further development to improve thruster lifetime will go through significant development and testing before integration into the existing cluster architecture. Estimates for the cluster mass are just over 15 kg using close to 16 W maximum during normal operation.

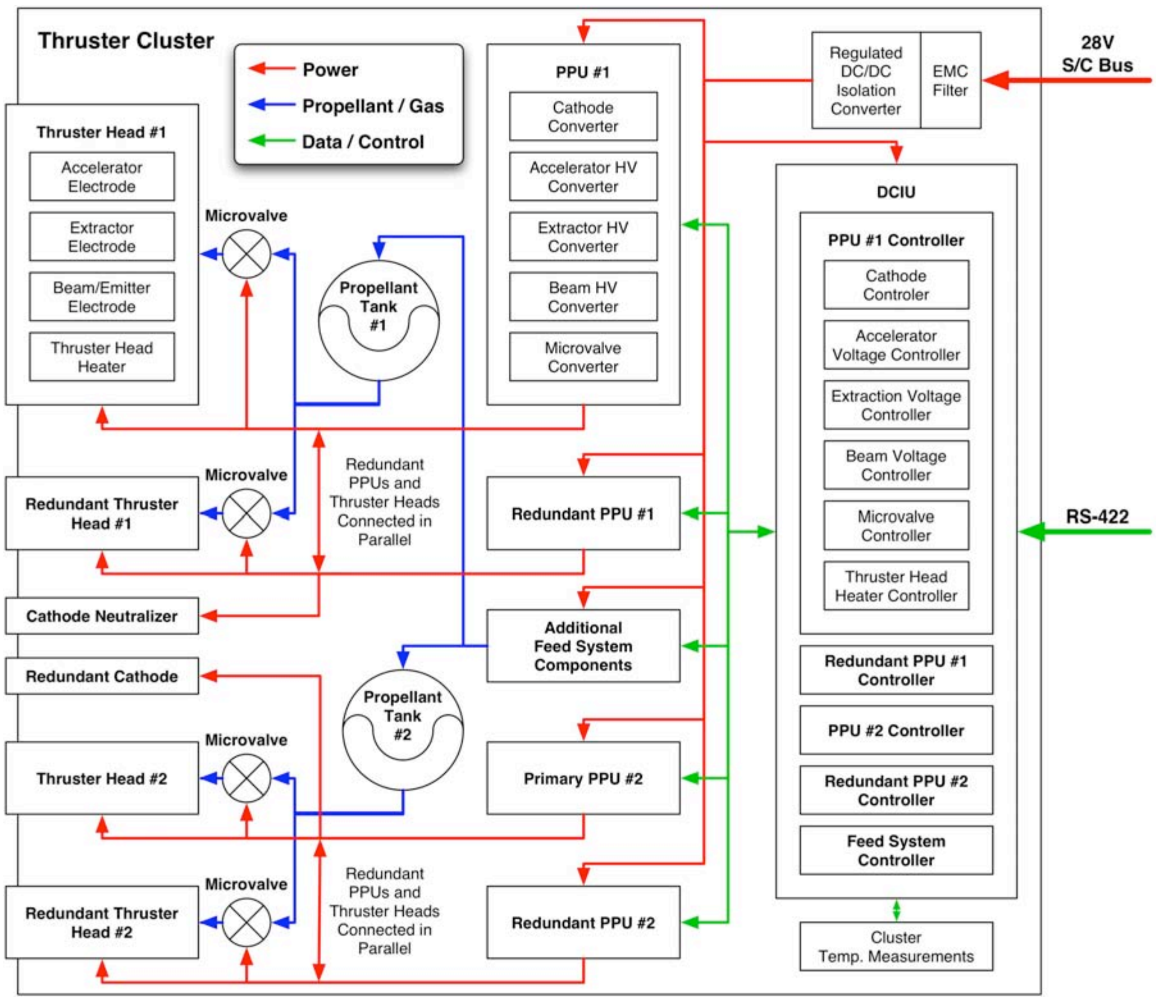


Figure 3.5-5: LISA CMNT Propulsion Architecture

3.6 Communication System Design

Mission baseline parameters driving the LISA Communication System design are referenced in Table 3-19.

Table 3-10: Communication System Baseline Parameters

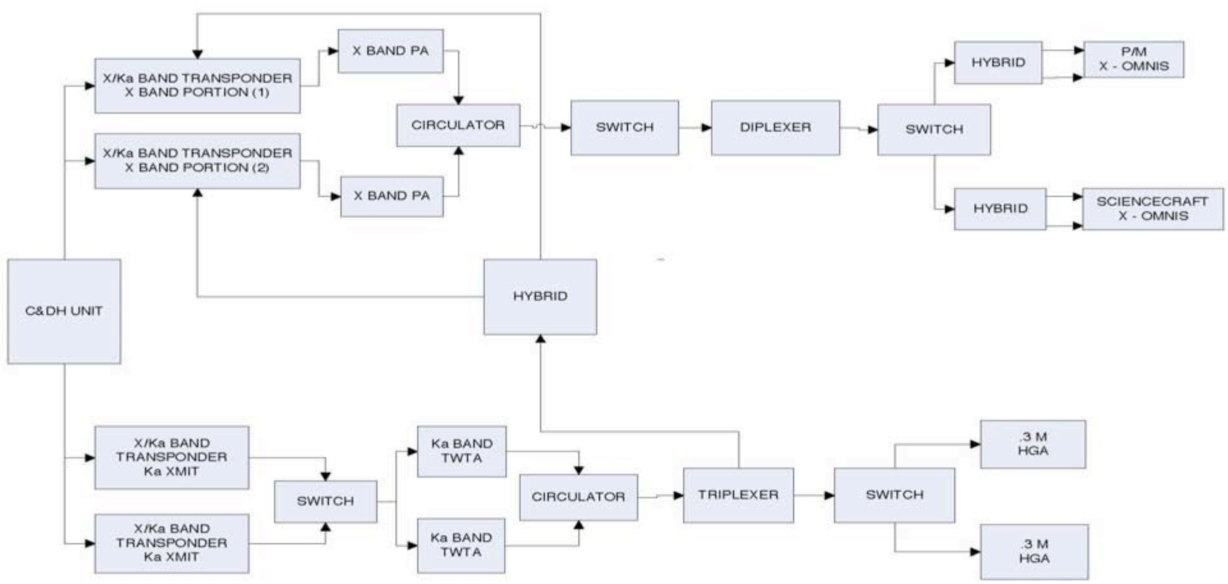
Parameter	Requirement	Comments
Lifetime	+ 5 year science operations phase (10 year goal) after 18 month cruise phase	14 months transfer trajectory + 4 months commissioning
Orbits	3 independent, Heliocentric, 20° earth trailing orbits, equilateral triangular constellation with 5 x 10^6 km +/- 1% arm lengths	Constellation requires no active station keeping or maintenance over the mission lifetime
Comm. Ground Link	8 hour DSN contact every 6th day for each S/C	DSN 34m dish is baseline X-band uplink Ka-band downlink
Comm. Data Rate	Minimum 5 kbps data rate per Sciencecraft for LISA science telemetry	
Data Delivery Efficiency	99% of data delivered to Science Operations Center	
Data Error Rate	10^{-6} bit error rate (BER) max.	

3.6.1 Communication System Overview

The LISA S/C Communication System will communicate with the Mission Operations Center (MOC) through the use of HGA and omni antennas operating on X-band and Ka-band frequencies. LISA science data and housekeeping data, estimated to require a data rate < 5 kbps per Sciencecraft, will be transmitted at 90 kbps on the Ka-band frequency via HGA antennas while command and S/C maintenance telemetry will be transmitted and received using both HGA and omni antennas on the X-band frequency. During the cruise phase, the Sciencecraft HGAs will be obstructed by the P/M shell, therefore omni antennae on the P/M will have to be used during the cruise phase. Both omni and gimbal mounted HGA antennae on the Sciencecraft will be available during the science phase. Communication on the ground will use a 34m DSN antenna. There will be no direct communication between the S/C except through the ground link.

3.6.2 Communication System Architecture

A schematic block diagram of the Communication System is shown in Figure 3.6-1. The system design provides optimal reliability through the use of redundant antennae, PAs, TWTAs and transponders as well as cross-strapping between the transponders, PAs, TWTAs and antennae. The Communication System controller card housed in the Sciencecraft C&DH unit, defined in greater detail in the C&DH section of this document, will handle all Communication System C&DH functions. Except for the antennas, all Communication System components for the Sciencecraft and Prop Module will be housed in a Communications Electronics Box mounted on the bottom deck of the Sciencecraft bus. A zero insertion force electrical connector built into the separation system will provide the electrical connection to the Prop Module omni antennae.



Source MDL Jan 08

Figure 3.6-1: S/C Communication System Block Diagram

3.6.2.1 HGA Installation

Two HGA gimbal units will be mounted on the solar array. Figure 3.6-2 shows approximate mounting locations for the HGA gimbal units.

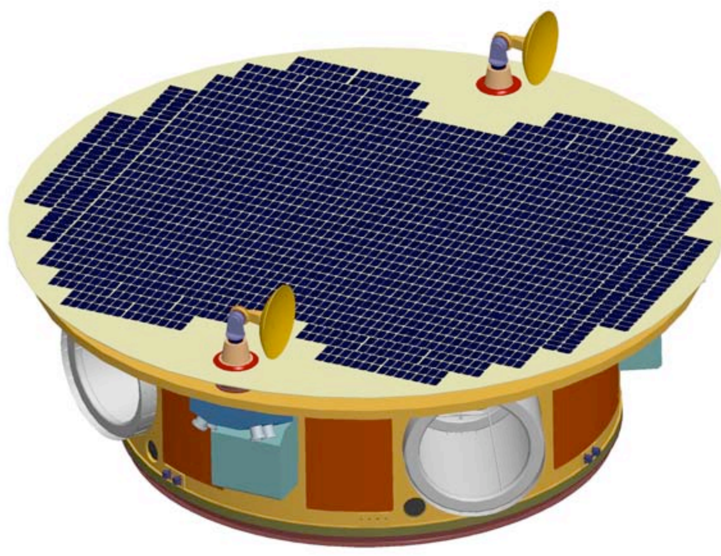


Figure 3.6-2: HGA Installation

3.6.3 COMMUNICATION MODES

A summary of the various communication modes is provided in Table 3-14. Contingency communication modes are available should a failure with the primary communication mode occur. The entire science data stream can be downlinked on the X-band frequency if required. The omni antennas will be incapable of transmitting science telemetry due to their limited data rate.

Table 3-14: LISA Communication Modes

Communication Mode	Mission Phase	Antenna	Freq.	Data Rate
S/C Telemetry to 34m DSN	Cruise	Omni	X-band	50 bps
34m DSN Command to S/C	Cruise	Omni	X-band	50 bps
Sciencecraft Science Data to 34m DSN	Science	HGA	Ka-band	90 kpbs
34m DSN Command to Sciencecraft	Science	HGA	X-band	2 kbps
Contingency Modes				
Sciencecraft Science Data to 34m DSN	Science	HGA	X-band	10 kbps
34m DSN Command to Sciencecraft	Science	Omni	X-band	50 bps

3.6.4 LISA Science Telemetry

The LISA constellation science telemetry data stream will consist of three components:

- Science data
- Science housekeeping
- Engineering housekeeping

The estimated total science telemetry data rate per Sciencecraft is 5 kbps continuous which includes science data and both science and engineering housekeeping. This assumption easily covers the expected science data rate of approximately 1 kbps and includes 15% overhead for coding and 25% contingency. The estimated data rate for science and engineering housekeeping is < 4 kbps.

3.6.4.1 Science Data

The science data consist of the decimation filtered phase-meter data (or linear combinations thereof) and the ancillary data as needed for arm length determination and clock synchronization.

Table 3-15 lists the science data continuously acquired from the LISA constellation which are needed as input to baseline ground processing. The numbers in parentheses show how the data rate might be reduced by some simple data processing on board and sending only differenced quad cell signals to the ground.

Table 3-15: Constellation Science Data

	Number of Signals in Constellation	Accuracy [bit]	Total Bits per sample	Total data rate @3 Hz sampling [bit/s]
High (3-main s_x, s'_x data steams dominated by a high dynamic range main detector)	6	24	144	432
Mid (3-main s_x, s'_x data steams dominated by bench noise, 3 $\tau_{x\_}$ data streams dominated by bench noise)	6	10	60	180
Low Pointing signals (differential quadrant signals)	36 (24)	10	360 (240)	1080 (720)
σ_i-s_i Signals (USO phase noise measurement)	6	19	114	342
Time semaphores	6	96	576	576 (@1 Hz)
Total Data Rate [bit/s] Entire constellation				2610 (2250)
Total Data Rate per Sciencecraft (bit/sec)				870 (750)

3.6.4.2 SCIENCE HOUSEKEEPING DATA

The Science housekeeping data consist of continuously measured environmental data (temperatures, magnetic field, proton radiation) and data of the sensors used in the DRS Controls. Both types of data are low-pass filtered and represented as a continuously sampled data stream at low bandwidth (typically <0.1 Hz). Science housekeeping data can be used to improve the main science data product by removing systematic errors or to flag time periods during which unusual environmental conditions are present. In general, the algorithms to be used for the utilization of science housekeeping will not be part of the baseline processing.

3.6.4.3 ENGINEERING HOUSEKEEPING DATA

Engineering housekeeping data consist of status data from Sciencecraft and payload technical equipment needed to identify malfunctions or parametric degradation. On board recording of housekeeping data (at increased average data rate) in a circular buffer capable of storing several weeks worth of data is foreseen to further enhance the usefulness of technical housekeeping for investigation of anomalies.

3.6.4.4 SCIENCE TELEMETRY QUALITY

Continuously acquired science data telemetry from the LISA constellation stored onboard each Sciencecraft will be downlinked to ground during scheduled DSN contact periods. The Communication System will ensure that 99% of the science data is received by the DSN with a bit error rate < 10^{-6} . This assumes clear sky coverate at one of the DSN sites.

3.6.4.5 GROUND LINK

Ground communication with the LISA constellation will be achieved using the DSN 34m dish as the baseline. Since the DSN antenna can have only one Sciencecraft in field of view at any time, it will be necessary to contact each Sciencecraft individually to downlink data telemetry. A baseline operational communications plan with the DSN contacting a different Sciencecraft every other day meets data latency requirements for ground processing. Each Sciencecraft HGA can be managed so that it is necessary to repoint once every 12 days, which meets the continuous data taking interval requirement. An eight-hour contact period for Ka-band communication is sufficient to downlink all of the data and remain within the capabilities of the DSN.

A comparison of the key uplink and downlink parameters for the DSN 34m dish is shown in Table 3-16 below. The parameters parameters represent average values assuming clear sky conditions at 20 deg elevation and are accurate within ± 2 dB.

Table 3-11: DSN 34m Key Uplink / Downlink Parameters

G/S Parameter		DSN 34m/X-band	DSN 34m/Ka-band
Antenna gain	Uplink	65.58	65.5
EIRP		120-130 dBm	138 dBm
Antenna Gain	Downlink	68 dBi	77-78 dBi
Effective G/T		53	60

3.6.5 Communication System Hardware

The two P/M omni antennas and respective hybrid unit and cabling are included in the list below. The components and performance specs listed below represent candidate hardware that might be used in the final LISA Communication System design. Hardware information accuracy will improve as the Communication System design matures.

3.7 Thermal Management System Design

Mission baseline parameters driving the LISA Sciencecraft Thermal Management System Design are referenced in Table 3-17.

Table 3-12: Sciencecraft Thermal Design Baseline Parameters

Parameter	Requirement	Comments
Cruise phase temperature	Temperature of all units on the SC and P/M (including payload) to be maintained within acceptable boundaries	The thermal subsystem must provide acceptable temperatures to all LCM elements throughout the cruise phase
On-station temperature	Temperature of all units on the SC within acceptable boundaries	The thermal subsystem must provide non-switching thermal control that maintains temperatures to acceptable limits
Thermal stability at payload interface during operational phase	10^{-6} K/√Hz @ 1 mHz TBR	Ensure thermally stable environment within critical parts of the payload during the operational phase

3.7.1 Thermal Management System Design Overview

The function of the LISA Sciencecraft thermal design is twofold; to protect the payload and mission critical components from the harsh thermal environment of space throughout the mission, and to maintain thermal stability within the LISA data bandwidth during the science phase of the mission. Survival of

Sciencecraft instruments during the cruise phase will be achieved through the use of heaters and MLI on the Sciencecraft and propulsion module. Active thermal control will be applied using thermostats and thermistors located inside the electronics boxes during the cruise phase. During the science phase, the survival heaters will be turned off, being supplemented by electronics box waste heat. Current thermal analysis indicates that the required thermal stability during the science phase can be achieved without the use of active temperature control.

During the science phase, the orbital configuration of the LISA constellation provides a thermally benign environment for the Sciencecraft. The near constant orbiting distance from the sun and the constant angle between the spacecraft normal vector and the solar vector provide near constant solar input. The distance from the earth serves to eliminate any significant effects from albedo or planetary heat sources. The remaining thermal disturbances are minimized through the use of passive techniques such as power stabilization for electrical components, low conduction isolators and radiation shielding

3.7.2 Thermal Management System Design Architecture

A schematic of the Sciencecraft Thermal Management System Design is shown in Figure 3.7-1.

3.7.2.1 Layered Thermal Barrier Design

Thermal disturbance effects on the LISA Opto-mechanical Core System (LOCS) and LISA Instrument Metrology and Avionics (LIMAS) systems will be minimized by employing three layers of radiation and conduction isolation:

Layer 1: The Solar Array Deck (SAD)

The Solar Array Deck is isolated from the Bus structure using low conductivity flexures and low emissivity coatings.

Layer 2: The Payload Shield

The payload is mounted to the Bus structure using low conductivity mounts. Attached to this mounting frame is a thin hexagonal shield with low emissivity coatings to provide radiation isolation.

Layer 3: Payload Internal Shield

A secondary shield internal to the payload shield is also mounted with conductive isolators and low emissivity coatings.

3.7.2.1.1 Layer 1: Solar Array Deck (SAD)

The first layer of isolation is the SAD. The SAD will be made of aluminum or composite honeycomb. The sun facing facesheet of the SAD contains both gold or Vapor Deposited Aluminum (VDA) coated Optical Surface Reflectors (OSR) and solar cells to increase the radiative path to space for heat to be rejected. The top surface of the bus structure top deck panel will be gold coated to minimize absorption of any radiation from the bottom surface of the SAD. A high emissivity coating around the bottom outer edge of the solar array will facilitate heat rejection to space. This isolation results in less than 1% of incident solar energy being transferred to the structure.

3.7.2.1.2 Layer 2: Payload Shield

The second layer of isolation is provided by the Payload Shield, which is gold coated on the inside and outside surfaces in order to minimize the absorption of any radiated energy (e.g. electronic box radiated heat). Low conductivity standoffs are also placed between the Payload Shield and the bottom plate to reduce the conduction path.

3.7.2.1.3 Layer 3: Payload Internal Shield

The last layer of payload thermal isolation is provided by an Internal Shield, which is gold coated on both sides and includes low conductivity mounts. This helps to shield the optical bench from any disturbances or gradients in the Payload Shield.

3.7.3 THERMAL STABILITY DESIGN

In addition to the layered thermal isolation design described above, the following design features will be employed to ensure compliance with the thermal stability science requirement:

- Use of low conductivity interface material, such as Choseal©, at all bus/payload interfaces
- Use of low conductivity fasteners, such as G-10, for all components requiring isolation
- Isolate and insulate battery from bus interior using a low conductivity interface and thin film radiation enclosure
- Isolate X-band antenna(s) from spacecraft using low conductivity material
- Use make-up heaters to simulate KA-band transponder load to ensure constant thermal environment
- Mount high heat dissipating boxes to bottom deck only and avoid mounting boxes to top deck
- Apply high emissivity Aeroglaze© Z307 electrically conductive black paint to Sciencecraft bus interior to draw heat away from payload
- Apply high emissivity coating such as NS43G white paint on Sciencecraft bottom and side exterior surfaces to reject heat to space

3.7.4 THERMAL ANALYSIS

The current thermal models of the LISA Sciencecraft, shown in Figure 3.7-2, Figure 3.7-3, and Figure 3.7-4, indicate that the required temperature stabilities can be achieved and the optical bench will be near room temperature without any active temperature control[13]. While heaters may be present for safe mode, etc., the current baseline does not include any active temperature control. Open-loop control is used for all heating on the Sciencecraft. Thermistors are used for monitoring the temperatures of all units.

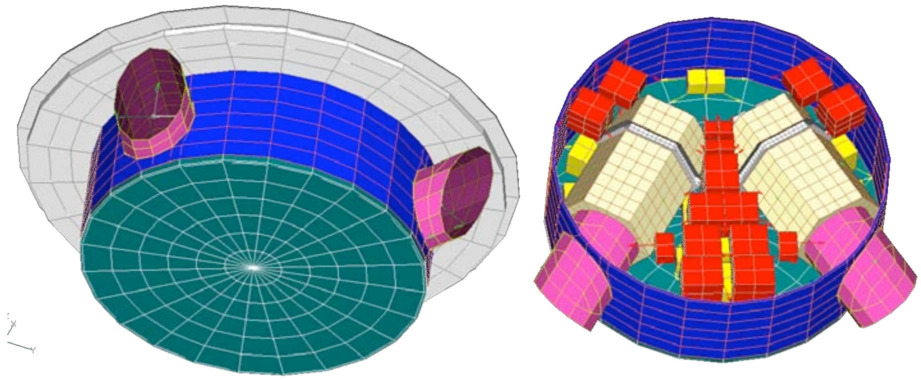


Figure 3.7-2: Sciencecraft Thermal Finite Element Analysis Mesh Definition

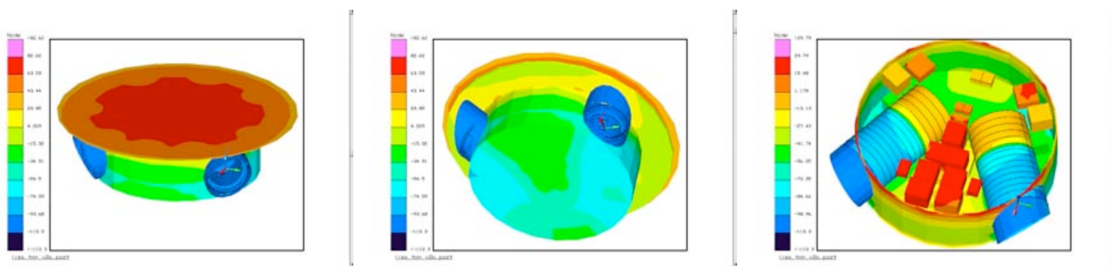


Figure 3.7-3: Steady-State Temperature Predictions

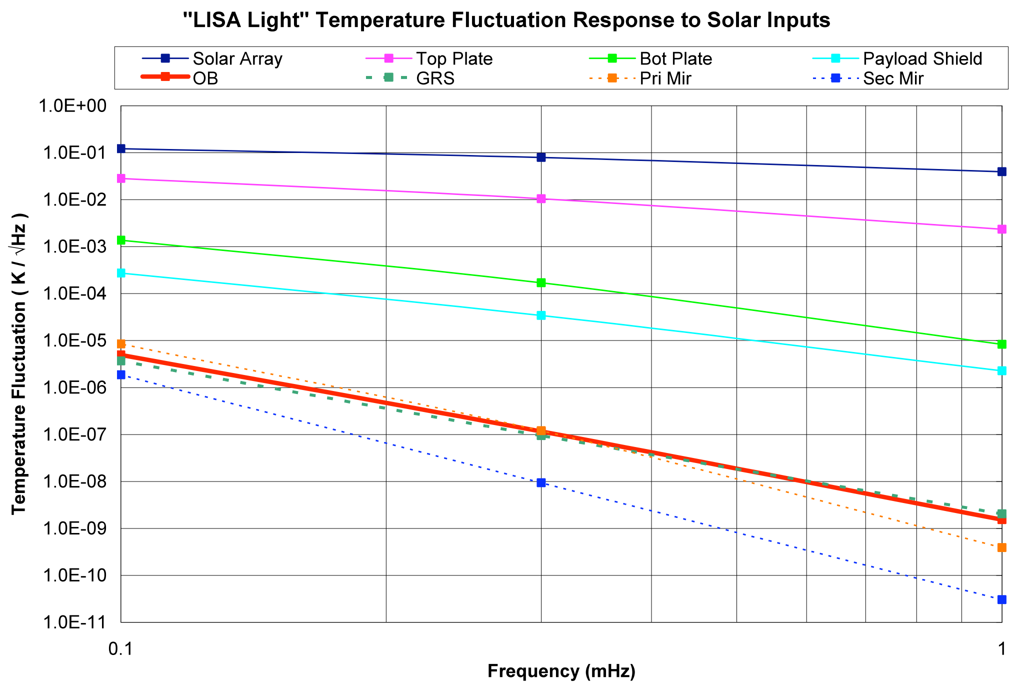


Figure 3.7-4: Temperature Fluctuation Predictions vs. Frequency for Solar Inputs

3.8 Flight Software Design

Mission baseline parameters driving the LISA Software Design are referenced in Table 3-18.

Table 3-18: LISA Flight Software Design Baseline Parameters

Parameter	Comment
Lifetime	+ 5 year science operations phase (10 year goal) after 18 month cruise phase (14 months for transfer trajectory + 4 months commissioning)
C&DH	Supports Sciencecraft functions only, science data processing is performed on the ground

3.8.1 Flight Software Overview

LISA flight software (FSW) hosted in the Main Command and Data Handling (C&DH) Processor will include onboard FSW to control the Sciencecraft, establish and maintain the laser links between

Sciencecraft, and provide data processing and delivery. Some aspects of the FSW, for example the DRS controls that manage attitude of the Sciencecraft, are integrated with the science data collection process. On-board science data processing is expected to be at a minimum, but it is necessary to do some processing to help assess the health and safety of various systems such as laser frequency or amplitude noise or the noise level in the main beat signal.

One of the key features of LISA's FSW is flexibility – from several perspectives. First, contact with the ground (via DSN) is expected to be only once every 6 days per Sciencecraft. That will require FSW that is capable of a moderate degree of autonomy. Specifically, avoiding large gaps in the data demands some capability to respond to at least minor error conditions without ground intervention. Second, the FSW responsible for data delivery should be capable of altering the contents of the data stream being sent to the ground (for example, via ground-modifiable filter tables and over-writable buffers) so that we can retrieve stored data or data that is routinely collected but not routinely sent to the ground, or perhaps data that needs to be relayed to the ground for one of the other Sciencecraft. Third, we will most likely want to include the ability to modify the FSW post-launch, both to fix problems that are discovered and to take advantage of new developments and algorithms. An available architecture to facilitate these capabilities is called the Core Flight System (CFS) technology currently under development at GSFC and expected to be demonstrated on the Global Precipitation Measurement (GPM) mission in 2013. A forerunner of the CFS, the Core Flight Executive (cFE) will fly on the Lunar Reconnaissance Orbit (LRO) mission expected to be launched in 2008. A discussion of LISA FSW architecture and the interplay of the cFE/CFS with the LISA-specific FSW mission applications are provided in the following subsections.

The hardware architecture that the FSW architecture will need to support is illustrated in the electrical system section 5.3 (ref Electrical Block Diagram). Major FSW components will reside in the Main C&DH processor and the LIMAS Phasemeter Front End Processor, with additional FSW elements probably present within the LOCS Laser Stabilization and the GRS Front End Electronics (FEE-SAU). Only the functionality of the FSW resident in the Main C&DH processor will be discussed in detail in this section. Although execution of many local functions can be carried out without sharing information between processor components (for example, charge management within the GRS will primarily rely on information sensed locally to determine when to the turn on its UV light), most onboard functions will require input from a different major area (e.g., the DRS housed in the C&DH main processor needs GRS data to perform drag-free control), or will provide output to a another major component (e.g., the LIMAS's Phasemeter output drives the synchronization of the LOCS's laser frequencies). To support this internal communication within a Sciencecraft, a simple on-board network and file system will be utilized. Some intra-Sciencecraft network, via a laser communications link using sidebands on the science laser link, is possible as well.

The FSW will have to manage multiple control loops. Within the DRS control loop, the LIMAS, GRS, and Attitude Control Subsystem (ACS) sensors will generate input data that will be used by the DRS to generate the micro-Newton thruster and proof mass commanding necessary to maintain drag-free control. The LIMAS will also generate the input required by the LOCS to maintain the required laser frequencies during science operations. To achieve the laser signal acquisition for all arms of the constellation required for science operations to commence, the Sciencecraft ACSs will drive scanning motions using input from the acquisition CCDs. Each of these major control loops (and others) may have their own processing rates and may be implemented as separate FSW tasks controlled by the main processor's executive functions. FSW complexity is influenced by several factors such as the number of operating "modes" of a Sciencecraft, the attitude and control requirements, autonomy requirements, and the reliability/redundancy of components. The LISA Mission is unique in that there are 3 Sciencecraft that are dependent upon each other to support the science, so in a sense, the “Instrument FSW” is actually spread across the 3 Sciencecraft. Complexity is somewhat reduced since the FSW is identical on all three Sciencecraft. The attitude and control algorithms are well-developed and understood (based on algorithms already developed for LISA Pathfinder), the science calculations are performed on the ground, and the Sciencecraft FSW is based on a layered architecture and core product used on current NASA missions. This product and architecture allows the FSW development to be concentrated on mission unique requirements. Reliability is almost inherently built into the science as there are 6 “legs” collecting science, 4 (TBR) of which are minimally required to meet mission needs. Redundancy is dependent on the risk tolerance of the mission managers and engineers, as well as cost. There must be some autonomy onboard since any given Sciencecraft may be out of contact with the ground for days (TBR).

3.8.2 LISA FSW Architecture

3.8.2.1 Generic FSW Architecture

Figure 3.8-1 shows an example of the generic FSW architecture that would define the infrastructure housing the LISA FSW applications. The architecture uses a layered design where higher layers can use services defined by lower layers. Layers support on-orbit application modification because they can insulate lower layers from changes in higher layers. Successful isolation depends on the functionality contained within each layer and how the layer’s interface is controlled. The layers defined in Figure 5.8-1 have evolved over a decade of GSFC Flight Software Branch (FSB) missions. This experience instills confidence that the architecture allocates the appropriate functionality to each layer. From a LISA subsystem perspective, layers impact what is configuration managed as a distinct component (i.e., LOCS, LIMAS, ACS, EPS, Comm, Prop, etc.), how the components are related, and what supporting artifacts (for example, software tools, test sets, files, etc.) are required. The top layer (Mission Support, ACS, and Instrument Support) is where the LISA-specific applications will reside. The lower layers vary depending on the avionics, OS, and core functionality required.

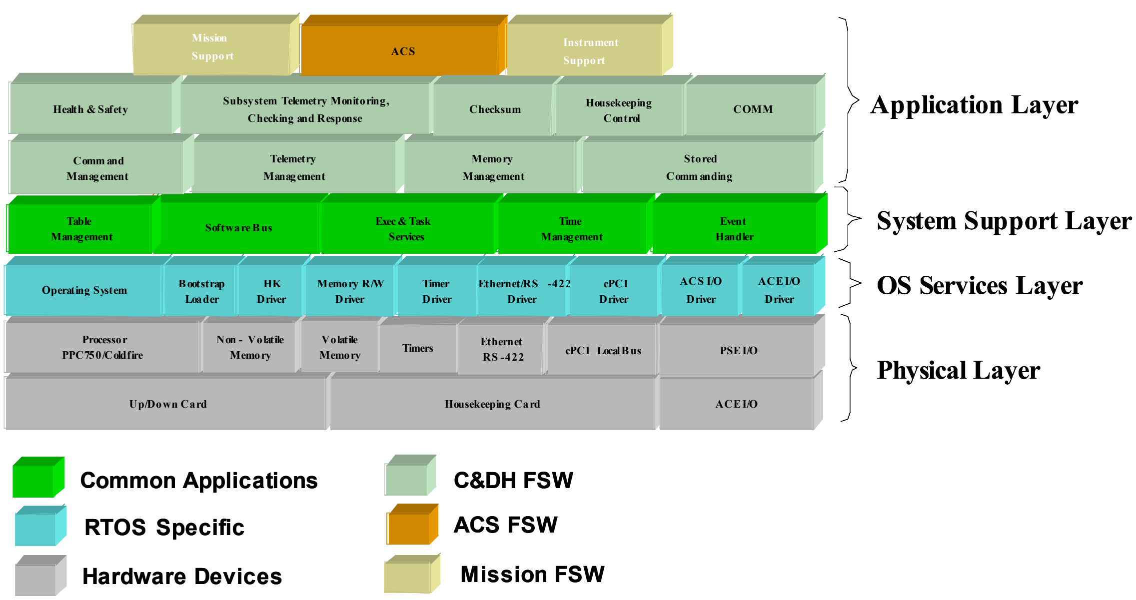


Figure 3.8-1: Generic FSW Architecture

3.8.2.2 Core Flight Software (CFS) Architecture

Figure 3.8-2 is a simplified illustration of the CFS FSW architecture, where the detailed functional breakdown of Figure 3.8-1 has been grouped into larger functional areas to show the relationship between the hardware-specific, CFS, and LISA mission-specific applications more clearly. This diagram shows how most of the generic FSW support functions have been reorganized into the reusable cFE. The platform layer is subdivided into purely hardware elements (Figure 3.8-1's physical layer) and the software associated with flight hardware elements (the Boot software, hardware drivers, and operating system). In addition, it contains software used to interface with the cFE FSW, the part of the CFS that forms the backbone from which the FSW applications execute. Figure 3.8-2 also distinguishes between those elements that must be customized for the LISA mission, those that are commercial off-the-shelf (COTS) purchases, and those that may be reusable cFE components. Finally, Figure 3.8-2 shows how the cFE encompasses what Figure 3.8-1 identified as the System Support Layer, while the Application Layer in Figure 3.8-1 is now subdivided into reusable CFS elements and LISA mission-specific applications like Guidance Navigation and Control (GN&C). LISA will have many more elements, but for simplicity of illustration only the GN&C applications are explicitly shown.

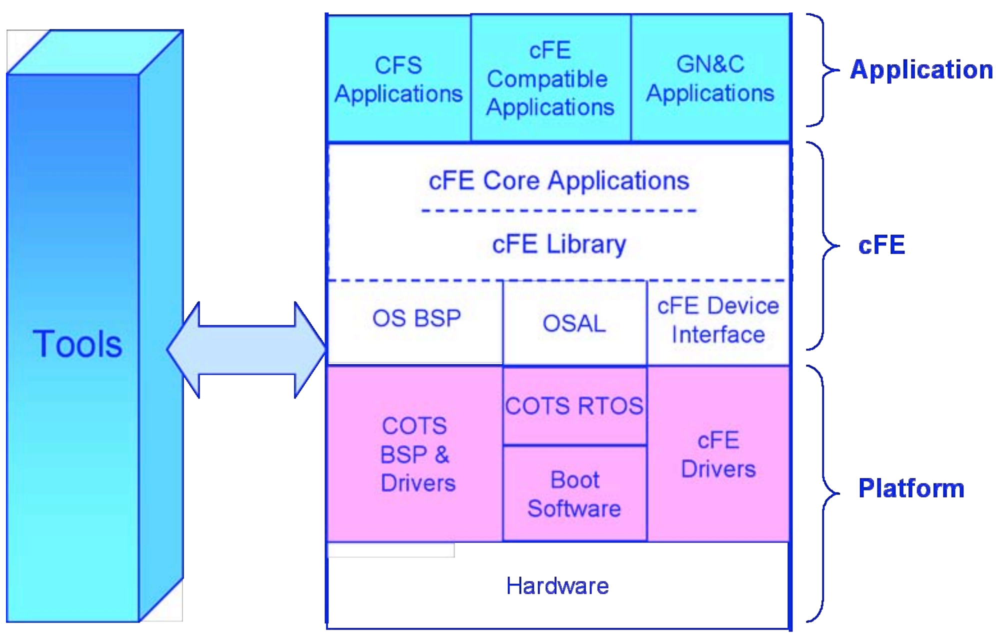


Figure 3.8-2: CFS FSW Architecture

Figure 3.8-3 restores the detail of Figure 3.8-1 in displaying the functional contents of the cFE. Further discussion of CFS layering structure and the individual functions implemented in the layers is provided in the following subsections.

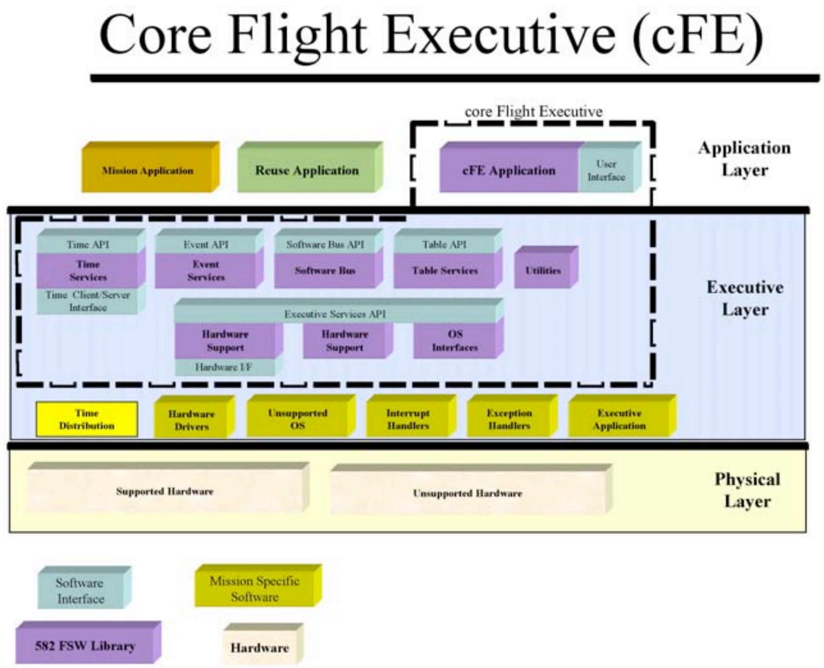


Figure 3.8-3: FSW Functions Contained within the cFE

3.8.2.3 PLATFORM LAYER

As its name implies, the Platform layer has dependencies on the hardware platform. Note that the term COTS software is being used in a general sense and includes open source as well as commercial software. Boot software executes when a processor is powered on. The Boot software works with the COTS Board Support Package (BSP) and device drivers to configure the hardware into a known state and to load the COTS Real-time Operating System (RTOS). The COTS BSP performs the following three functions: initialize the hardware platform to a known state, interface to basic hardware devices (i.e. drivers), and interface between the hardware platform and the COTS RTOS. The RTOS performs pre-emptive multi-tasking, interrupt and exception handling, and file system services. It also handles Resets, provides a debug monitor, and supplies the Bootstrap loader for performing memory loads and dumps. Note also that there are cFE components (Boot, BSP, & Device Interfaces) that may be tightly coupled with the platform layer that are configuration managed by the cFE.

3.8.2.4 CFE LAYER

The cFE layer contains two products: the cFE (which includes the Operating System (OS) BSP and the cFE Device Interfaces) and the Operating System Abstraction Layer (OSAL). The cFE is LISA-independent FSW that provides a runtime environment and a set of services for hosting FSW Applications. The cFE Library defines an Application Programming Interface (API) that remains constant across different hardware and/or OS platforms. The cFE Core Applications provide a ground interface to the cFE services. The OSAL is an open source product that provides an API to an abstract RTOS. The OSAL is hosted on both COTS real-time and non-real-time operating systems. The cFE Device Interfaces provide an API to abstract device interfaces. The software drivers to specific hardware are implemented by the cFE Device Drivers, COTS Device Drivers, or a combination of the two.

The cFE also includes the traditional System Management functions namely

- tables and file loads, dumps, and patches
- code loads, dumps, and patches
- memory integrity software

Specifically with respect to table services, the cFE provides simplified table management by eliminating the need for applications to contain code for managing their own tables. Tables can be shared between applications, but table data integrity is ensured by performing all table updates synchronously with the application that owns the table.

The cFE is responsible for communication within a Sciencecraft. The Software Bus provides a high-level mechanism for inter-task communication. It routes messages to all applications that have subscribed to the message. The subscriptions are done at the startup of an application and message routing can be added or removed at runtime. Routing subscriptions can be modified after launch by ground command or autonomously, but the application in question must have been originally implemented with the capability to accept and act upon the command. Of coarse, if not implemented pre-launch, a modified version of the application that incorporates this capability could be uplinked after launch. The Software Bus reports errors detected during message transference and outputs a Statistics Packet and Routing Information when commanded.

The cFE provides Time Services, including

- a user-interface for correlation of Sciencecraft time to the ground reference time (epoch)
- calculation of Sciencecraft time, derived from mission-elapsed time (MET), a Sciencecraft time correlation factor (STCF), and (optionally) leap second capability
- providing a functional API for cFE applications to query the time
- distribution of a “time at the tone” command packet, containing the correct time at the moment of the 1 Hz tone signal
- distribution of a “1 Hz wakeup” command packet
- forward tone and time-at-the-tone packets

The cFE does not interface directly with LISA’s USO. That interface is supplied via a separate hardware driver, which supplies a packet to the cFE which the cFE can process and reformat as discussed above.

The cFE provides Event Services. It provides an interface for sending asynchronous information or error messages in telemetry to the ground including

- a processor unique Software Bus event message containing the processor ID, Application ID, Event ID, timestamp, and the request-specified event data (text string including parameters)

- the ability to send messages via hardware message ports
- the ability to send long or short message format

The cFE can also store the asynchronous information/error messages in a buffer to enable downlink at a future time.

The cFE provides an interface for filtering event messages, and an applications event filtering that can be enabled/disabled. The cFE can also log the event in a local event log. These Event Services are used to support Health and Safety Management in the form of Telemetry and Statistics Monitoring.

Finally, Figure 3.8-4 below illustrates how the various cFE and CFS functions interact together via the Software Bus. The LISA-specific Applications are included at a somewhat simplified level. A more detailed discussion of these applications will be provided in the next subsection. Note that the data input to the ACS and DRS for attitude and antenna control is obtained from the GRS, attitude sensors, and HGA via the C&DH Data Acquisition Manager function. The commands to the GRS, Telescope Articulation, Point-ahead Actuator, Fiber Launcher, Prop Module Separation Mechanism, thrusters (micro-Newton thrusters when on station, hydrazine & biprop systems to get on station), and HGA are generated by the C&DH Command Generator function based on inputs computed by the DRS and ACS.

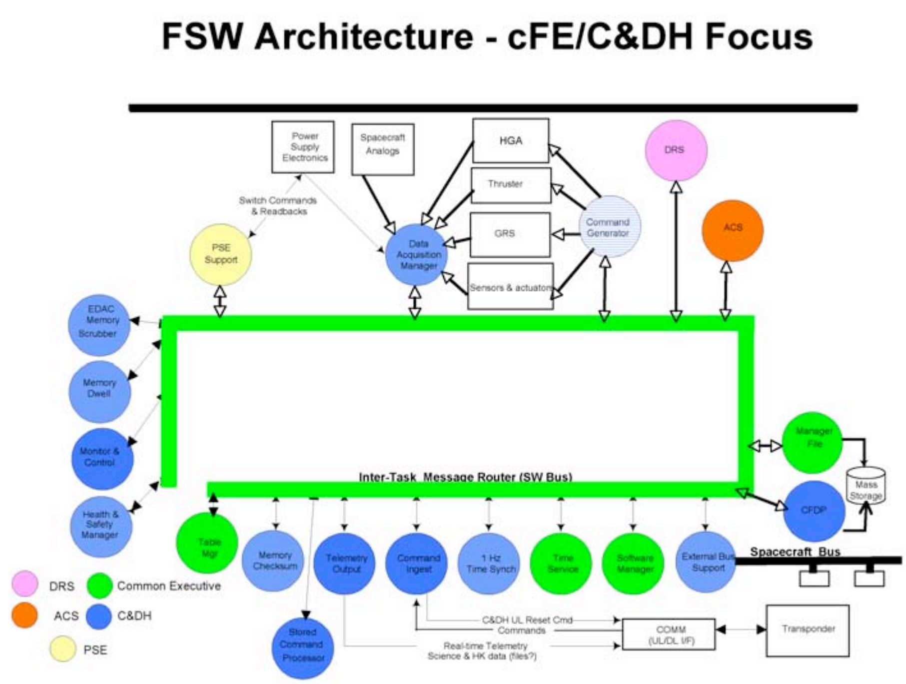


Figure 3.8-4: LISA FSW Architecture with cFE/C&DH Focus

3.8.2.5 GENERIC APPLICATION LAYER

A concerted effort was made to keep the cFE services to a minimum, while achieving the goal of providing a complete operational environment for applications. Minimizing cFE's functions and complexity maximizes the range of target platforms. Additional functionality can be selectively added, as needed, in the LISA Application layer.

Similar to the cFE, a core set of CFS Applications define CFS services. However, the cFE uses a library approach (i.e. function level API) while the CFS services use a Software Bus message-based interface. Onboard Command and Telemetry Management is an example of a CFS service. In the commanding area, the CFS handles command distribution and processing. The CFS provides autonomous onboard commanding with absolute and relative time-tagged sequences. In the telemetry area, the CFS collects and distributes Sciencecraft and instrument housekeeping data.

cFE compatible applications are applications that can run on the cFE, but are not included in the CFS. There are a variety of reasons why an application may be written this way. For example, an instrument may not need the CFS but would like to use the cFE. Therefore its applications would be cFE compatible, but not part of the CFS.

The final category of applications is the Mission-specific Applications. In a LISA context, the applications hosted in the Main C&DH Processor include the ACS, GRS management and control, propulsion, communications, power, etc. These LISA-specific applications will be compliant with the CFS architecture, but will be managed differently than CFS Applications. Figure 3.8-5 shows the generic Mission-specific FSW Framework. Similar to the overall FSW architecture, the Mission-specific framework uses layering. As with Figure 3.8-2, the boxes labeled "GN&C" serve as a token for the much greater number of LISA mission application classes.

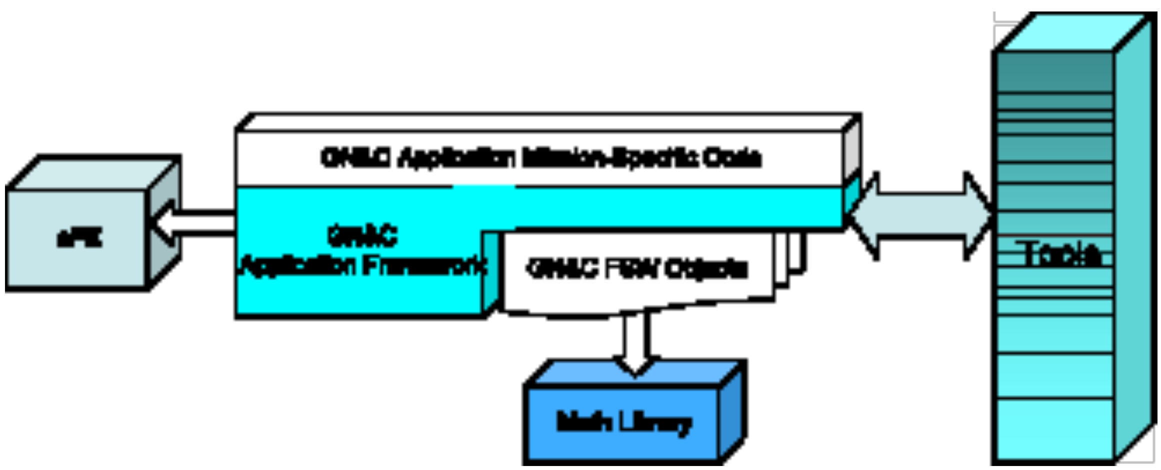


Figure 3.8-5: CFS Compliant Application Framework

The Math Library includes a collection of common math functions typically used by ACS FSW functions. Additional functions may need to be added to support the LISA mission. The Math Library is maintained as a separate product. Conceptually it can be thought of as the lowest architectural level, because it doesn't have any dependencies other than the ANSI C standard types.

The next higher level contains the Mission-specific FSW Objects. There are two types of Mission-specific FSW Objects: framework-dependent and framework-independent. Framework-independent objects can at most have dependencies on the Math Library and ANSI C libraries and they must follow some basic coding standards for items such as parameters that are stored in tables. The analytical Solar and Lunar models are examples of framework-independent objects. The significance of identifying framework-independent objects is that they can easily be used in other system such as an ACS analyst's simulators. Framework-dependent objects are Mission-specific FSW Objects that have one or more dependencies on the Mission-specific Application Framework. An example of a framework-dependent object is an object that accepts and validates an ephemeris Extended Precision Vector (EPV) from the ground. This object can be maintained separately from the ephemeris propagator which would most likely be a framework-independent object.

Encapsulating the Mission-specific FSW Objects is the Mission-specific Application Framework. This is a collection of utilities that is configuration managed as a single item. These utilities provide standard mechanisms for performing common application functions. For example a cFE interface utility provides the capability to register message callback functions for a cFE message pipe so that when a particular message is received the callback function is automatically executed.

The Mission-specific Application Code provides the "glue code" that configures the Mission-specific Application Framework for the LISA mission. For example, currently the ACS FSW Mission-Specific Application Code is manually written, but it is anticipated that a tool could generate most or all of this code.

3.8.2.6 LISA-SPECIFIC APPLICATION LAYER

Figure 5.8-4 illustrates how the various LISA-specific FSW Applications are tied into the overall FSW Architecture, along with the cFE and CFS functionality that have already been discussed in the previous subsections. The ACS and DRS functions together generate all the actuator control inputs. Specifically, they generate the desired HGA torque inputs, desired thruster firing inputs, desired telescope articulation changes, desired point-ahead actuator changes, fiber launcher & P/M separation mechanism command inputs, and desired proof mass suspension torques & forces. The ACS and DRS process data from the GRS and metrology subsystems (which supply the positions and orientations of the proof masses), star trackers, and CSSs to determine the desired thruster firing inputs and proof mass suspension torques & forces. Specifically, the DRS can use both the metrology function output from the Payload Processor and the capacitance sensing data output directly by the GRS to control the proof mass position and orientation. Proof mass charge control is also performed. Desired HGA torques are determined from ephemeris and current HGA gimbal data, while desired telescope articulation changes are determined from ground-specified commanded positions and current telescope articulation position data. Values for telescope articulation (and point-ahead actuator) are ground-determined for an extended time duration (probably specified via an uplinked table) and are controlled and implemented via the Telescope Articulation function (and Point-ahead Actuator function). Laser selection is also specified by the ground and implemented by the Main C&DH Processor FSW via commands to the Fiber Launchers.

Grouping the functionality (both custom LISA functionality and traditional GN&C functionality) into major FSW functional categories, the FSW resident in the Main C&DH Processor will

- Perform ephemeris modeling
- Process sensor and actuator data
- Collect housekeeping data associated with LISA-specific applications
- Process commands associated with LISA-specific applications

- Perform state estimation processing
  - Determining attitude and gyro drift biases from input of star tracker, Coarse Sun Sensor, and gyro data while attached to the P/M
  - Determining orientation and position (relative to other Sciencecraft) from input of GRS and star tracker data while acquiring science configuration and while performing science
- Manage and execute control modes
- Generate commands associated with LISA-specific applications
- Perform fault detection and correction (FDAC) associated with LISA-specific applications

In addition, the Main C&DH Processor FSW supports the power subsystem through its command & telemetry functions and FDAC. Specifically, the FSW processes PSE commands and collects power-related housekeeping telemetry. The Main C&DH Processor FSW also executes the solar array module control loop, and provides output module and Low Voltage Power Converter (LVPC) switches commands.

3.9 COMMAND AND DATA HANDLING (C&DH) SYSTEM DESIGN

Mission baseline parameters driving the LISA C&DH System are referenced in Table 3-20.

Table 3-20: C&DH System Baseline Parameters

Parameter	Requirement	Comments
Lifetime	+ 5 year science operations phase (10 year goal) after 18 month cruise phase	14 months transfer trajectory + 4 months commissioning
Bus voltage	28 V ±0.14 V.	Voltage supply to the payload, Bus voltage regulation will be 28 V ±2 V (TBR)

3.9.1 C&DH SYSTEM OVERVIEW

The C&DH system (a.k.a. Avionics System) approach for LISA reuses existing NASA developed designs and industry standards as much as possible. In addition, modularity in the design approach will be utilized. The C&DH System Architecture at a top level will:

- Host the LISA flight software
- Perform all C&DH functions: command ingest and distribution, telemetry distribution
- Provide Data Bus as required: low, high, or a combination based on speed required

3.9.2 C&DH System Architecture

A functional schematic of the Sciencecraft and the interfaces to the C&DH System is provided in Figure 3.9-1.

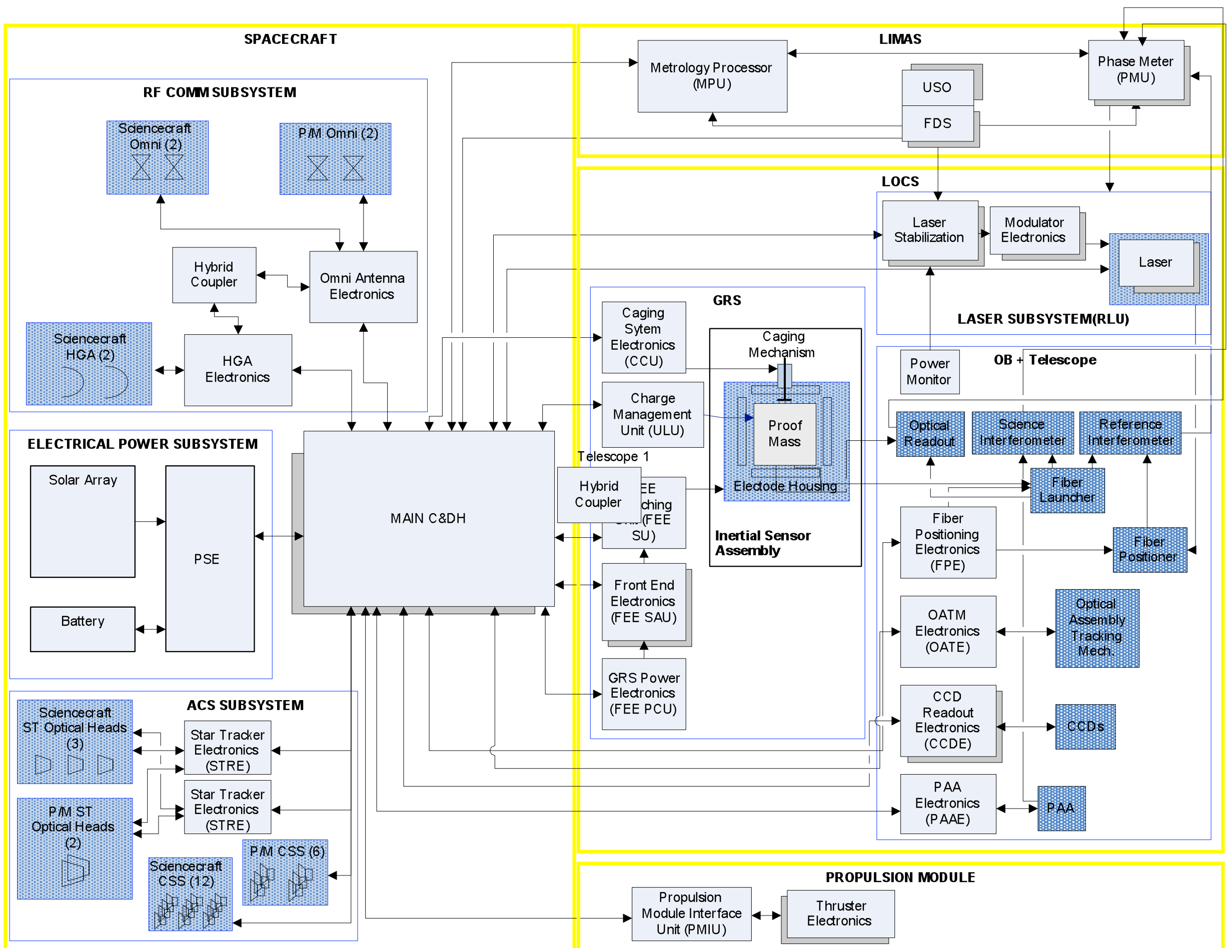


Figure 3.9-1: Sciencecraft Functional Schematic

3.9.2.1 Main C&DH Unit

A circuit board diagram for the main C&DH unit is shown in Figure 3.9-2. A prime and redundant processor will be housed in separate chassis, each providing full functionality to all the major subsystems as well as the payload. Multiple, flight proven, processors are available to meet the LISA requirements including: Rad750, and the GSFC cold fire processor. Each C&DH chassis will also contain a communications card, low voltage power card, housekeeping IO card(s), single board computer, multi-function analog card, a backplane, a USO, and most likely some type of mass storage. The data bus will either be a MIL-STD-1553, Spacewire, or both depending upon required speed (a high and low speed bus may be required).

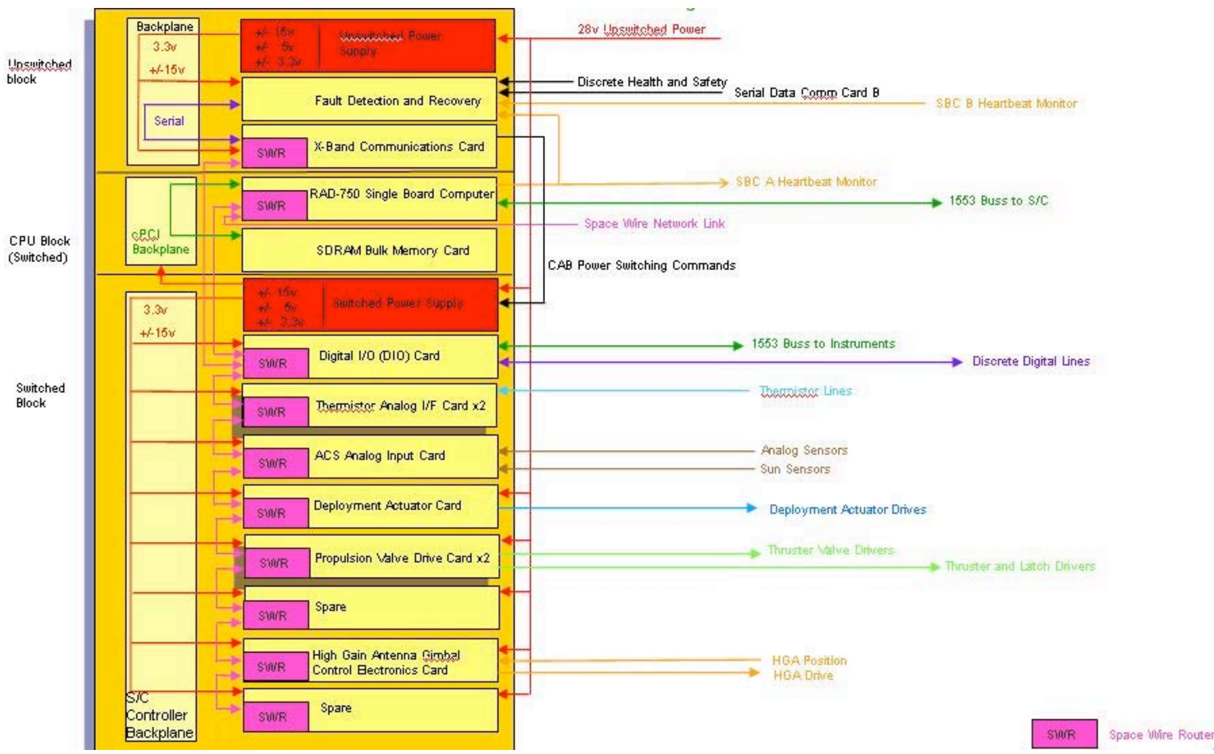


Figure 3.9-2: Main C&DH Unit Circuit Board Diagram

3.9.2.1.1 Main C&DH Unit Design

The internal components that comprise the main C&DH unit will be divided into the following three groups:

Group 1: Unswitched Power Block

Group 2: CPU Power Block (CPB), Switched

Group 3: Spacecraft Interfaces Block (SIB), Switched

The following Group 1 components make up the UPB:

1. Cold/Fire Based Fault Detection and Recovery Controller
    - Monitors state of CPU Block A, CPU Block B, alternate CAB, UPB
    - Primitive spacecraft control
2. X/Ka Band communication card

The following Group 2 components make up the CPB:

1. RAD750 based single board computer

2. Bulk memory card

The following Group 3 components make up the SIB:

1. Digital I/O card
2. Thermistor cards (2)
3. ACS analog sensor input card (Coarse Sun Sensors)
4. Deployment actuator driver card
5. Propulsion valve drive and latch cards (2)
6. HGA controller card

3.9.2.1.2 Main C&DH Unit Size and Mass

Dimension and mass estimates for the UPB block of the main C&DH unit are provided in Figure 3.9-3.

Integrated Avionics Chassis Mass/Size Calculator

Number of C&DH Cards	3	
Number of Controller cards	0	
Redundant Configuration (yes/no)	yes	
Integrated Configuration (yes/no)	no	
Size (3u/6u)	3u	
Wall thickness	100	mills
Partition Thickness	100	mills
Height	185.0	mm
Depth	150.8	mm
Length	111.8	mm
Ends	141722	(mm)^3
Sides	105032	(mm)^3
Top and Bottom	85615	(mm)^3
Card Guides	105032	(mm)^3
Partition	52516	(mm)^3
Divider	0	(mm)^3
Total Volume	489917	(mm)^3
Chassis Mass	1.3	kg

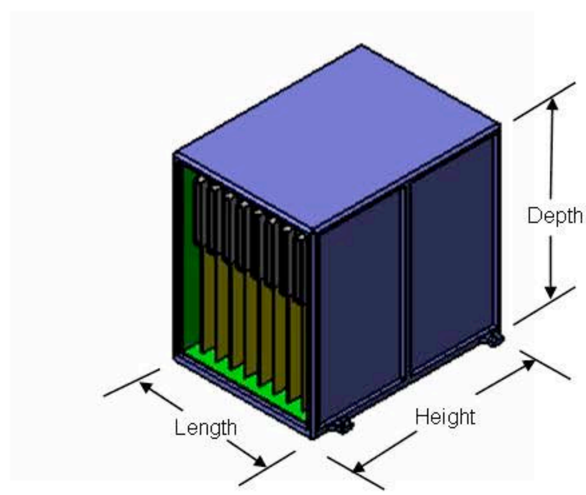


Figure 3.9-3: Main C&DH Unit UPB Block Size and Mass Estimates

3.9.2.1.3 Main C&DH Mass and Power Summary

A breakdown of the mass and power requirements for the UPB, SIB and CPB blocks of the main C&DH Unit are provided in Tables 3-21, 3-22, 3-23 and 3-24.

Table 3-21: Main C&DH Unit UPB Mass Breakdown

Table 3-22: Main C&DH Unit UPB Power Breakdown

Assembly	Powered	Avg Power	Total Power	Peak Power
Fault Detection and Recovery	2	4	8	4
S-Band Communications	2	6	12	6
LVPS	2	2	4	2
Chassis	0	0	0	0
Total			24	12

Table 3-23: Main C&DH Unit SIB/PCB Mass Breakdown

Circuit Board	Strings	N/String	Mass	Total Mass
RAD 750 SBC	2	1	0.8	1.6
Bulk Memory Card	2	1	0.8	1.6
LVPS	2	1	0.8	1.6
Digital I/O	2	1	0.8	1.6
Thermistor Analog I/F Card	2	2	0.8	3.2
ACS Analog Input Card	2	1	0.8	1.6
Deployment Actuator	2	1	0.8	1.6
Propulsion Valve & Latch Drivers	2	2	0.8	3.2
High Gain Antenna	2	1	0.8	1.6
Harness	2	1	0.2	0.4
Backplane	2	1	1.1	2.2
Chassis	1	1	3.2	3.2
Total				23.4

Table 3-24: Main C&DH Unit SIB/PCB Power Breakdown

Assembly	Powered	Avg Power	Total Power	Peak Power
RAD 750 SBC	1	7	7	7
Bulk Memory	1	6	6	6
LVPS	1	20	20	2
Digital I/O	1	4	4	4
Thermistor	2	4	8	8
ACS Analog Input	1	4	4	8
Deployment Actuator	1	0	0	10
Propulsion Valve Driver	2	0	0	40
High Gain Antenna Control	1	0	0	20
Backplane	1	0	0	0
Chassis	1	0	0	0
Total			49	105

3.9.3 C&DH Design Alternatives

3.9.3.1 Power System Electronics (PSE) Integrated into Main C&DH Unit

A design alternative that could yield mass and cost savings would be the integration of the Power System Electronics (PSE) box into the Main C&DH unit. A study of this alternative was conducted during the March 2008 GSFC Mission Design Lab (MDL) session. Refer to the Avionics final presentation from the March 2008 MDL session for more details.

3.9.4 C&DH Hardware Summary

Table 3-31 provides a Sciencecraft C&DH System hardware summary. The components and performance specs listed below represent candidate hardware that might be used in the final LISA C&DH System design. Hardware information accuracy will improve as the C&DH System design matures.

3.10 Redundancy Strategy

The spacecraft is designed to be at least single point failure tolerant for all credible failure points. This section briefly summarizes the design features of each spacecraft subsystem that ensure redundancy.

3.10.1 Electrical Power Redundancy

Both the solar array and the Li-ion battery include multiple redundant strings to enable non-critical degradation of the available power in the event of a single string failure. The solar array and battery are thus oversized to allow nominal power supply to be unaffected by a single string failure. The power system PCDU controller is internally redundant and able to tolerate the failure of a single internal module. The power regulation function is triple majority voted.

3.10.2 Command & Data Handling Redundancy

The on-board computer is fully internally redundant with redundant power supply units, mass memory units and processor modules. The I/O modules are also replicated to ensure redundancy in the command decoding function. The reconfiguration module and the command/telemetry outputs are also hot redundant.

3.10.3 Attitude Control System (ACS) Redundancy

Each Sciencecraft has 3 Star Trackers, two of which are aligned with each telescope, such that the failure of one Star Tracker aligned with each telescope can be tolerated (this provides single failure tolerance at the level of each telescope). In the event of multiple Star Tracker failures, the gyros could be used for attitude estimation. The gyros are also fully replicated, with hot redundancy of gyro channels (three provided in each unit), providing 6 skewed gyro axes to allow fault detection and isolation. The 12 Sciencecraft mounted Coarse Sun Sensors (CSSs) are used in a majority voting configuration providing active redundancy in the event of sensor degradation or failure. Redundancy is included in the design of the micro-Newton thruster configuration for both the colloid and FEEP options. Each colloid thruster contains a redundant thruster head and microvalve assembly while the FEEP design includes redundant thruster clusters.

3.10.4 Telecommunications Redundancy

The communication system is fully redundant with repeated transponders, TWTAs and antennas. Failure in any one of these elements will result in a fallback to the remaining unit. The high gain antennas operate in hot redundancy (both amplifiers are kept on in order to minimize thermal-cycling effects) and take turns to communicate with the Earth groundlink. This allows the frequency of steering events (which interrupt the science mode) to be halved. Redundancy is maintained, with the effect of one antenna failure being no loss in communication capability, but a doubling of the steering event frequency.

3.10.5 Thermal Redundancy

Over time the degradation of the surface finish on the spacecraft will reduce the ability to dissipate thermal energy, such that the spacecraft temperature is likely to rise over time. Sufficient design margin, layering of materials and careful material selection and quality control during manufacture will ensure that degradation of surface finish does not affect the broad thermal characteristics of the spacecraft to the extent where the mission performance is compromised. Also each heater circuit is controlled through majority voting of three thermistors that provides redundancy in the event of thermistor failure. Furthermore, each heater circuit is repeated.

4 LISA PAYLOAD

The following section presents an overview of the LISA payload design and its interfaces to the Sciencecraft. The purpose of this section is to provide a contextual reference for the Sciencecraft design. Detailed performance and interface requirement information for the LISA Payload can be found in the LISA Payload Description Document and the LISA Payload Interface Control Document respectively.

4.1 LISA PAYLOAD OVERVIEW

The LISA Payload is programmatically divided into two major functional blocks: the LISA Opto-mechanical Core System (LOCS) and the LISA Instrument Metrology and Avionics system (LIMAS). The LOCS includes the Laser Assembly, and the Opto-mechanical and structural components of the measurement system mounted in two movable Optical Assemblies (OAs), as shown in Figure 4.1-1 below. The LOCS also includes component drive electronics for the movable OAs. The LIMAS includes all the devices needed for the processing of the raw interferometer data and successive data handling (Phase measurement system). The LOCS and the LIMAS contribute together with the Sciencecraft ACS to meet the LISA performance requirements.

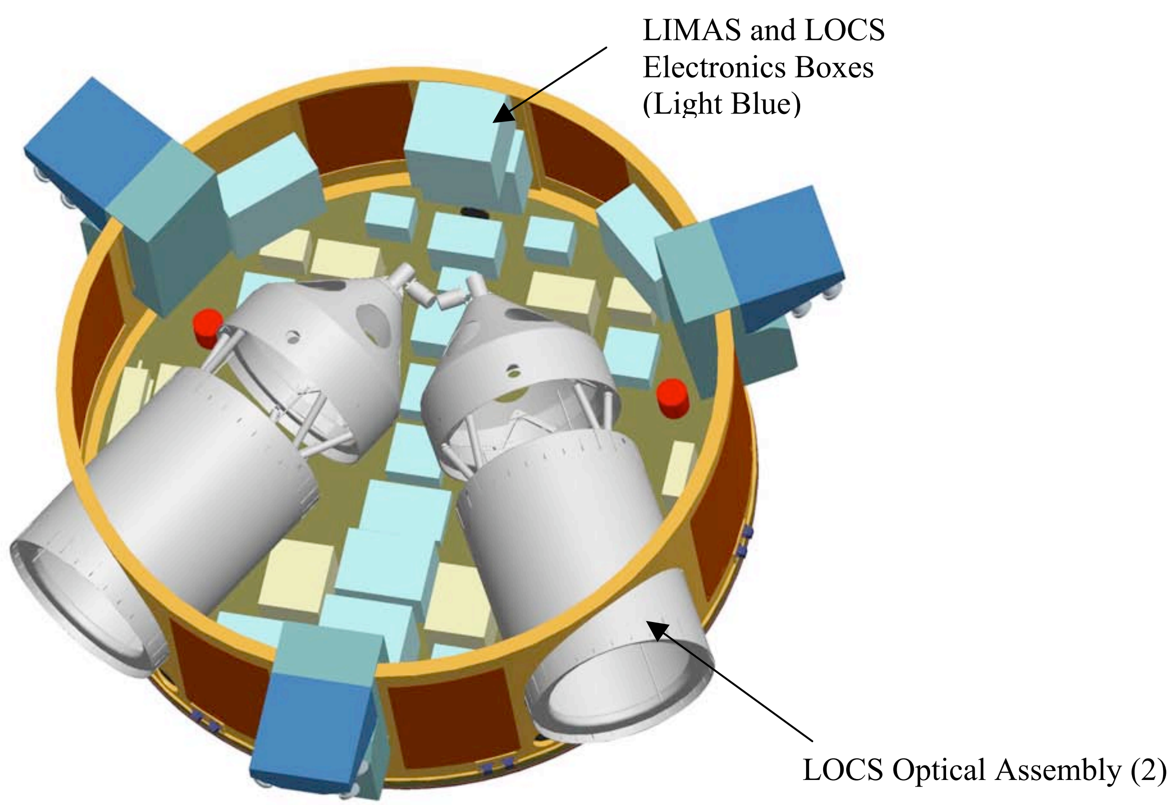


Figure 4.1-1: LISA Payload Overview

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4.2 PAYLOAD DESIGN AND INTERFACES

4.2.1 PAYLOAD DESIGN

The primary components of each LOCS assembly are the Gravity Reference Sensor (GRS), the Optical Bench, the Telescope assembly, the structural elements necessary to support and manipulate the Optical Assembly and the LOCS electronics box installation. The LIMAS basically consists of photodiodes and their trans-impedance amplifiers, the USO, the phasemeter front end electronics and a back end hosted in the payload metrology processor. The electronics boxes are designed to minimize radiated heat and EMI exposure. An illustration of one LOCS Optical Assembly, which consists of the optical bench, the telescope and the structural elements is provided in Figure 4.2-1.

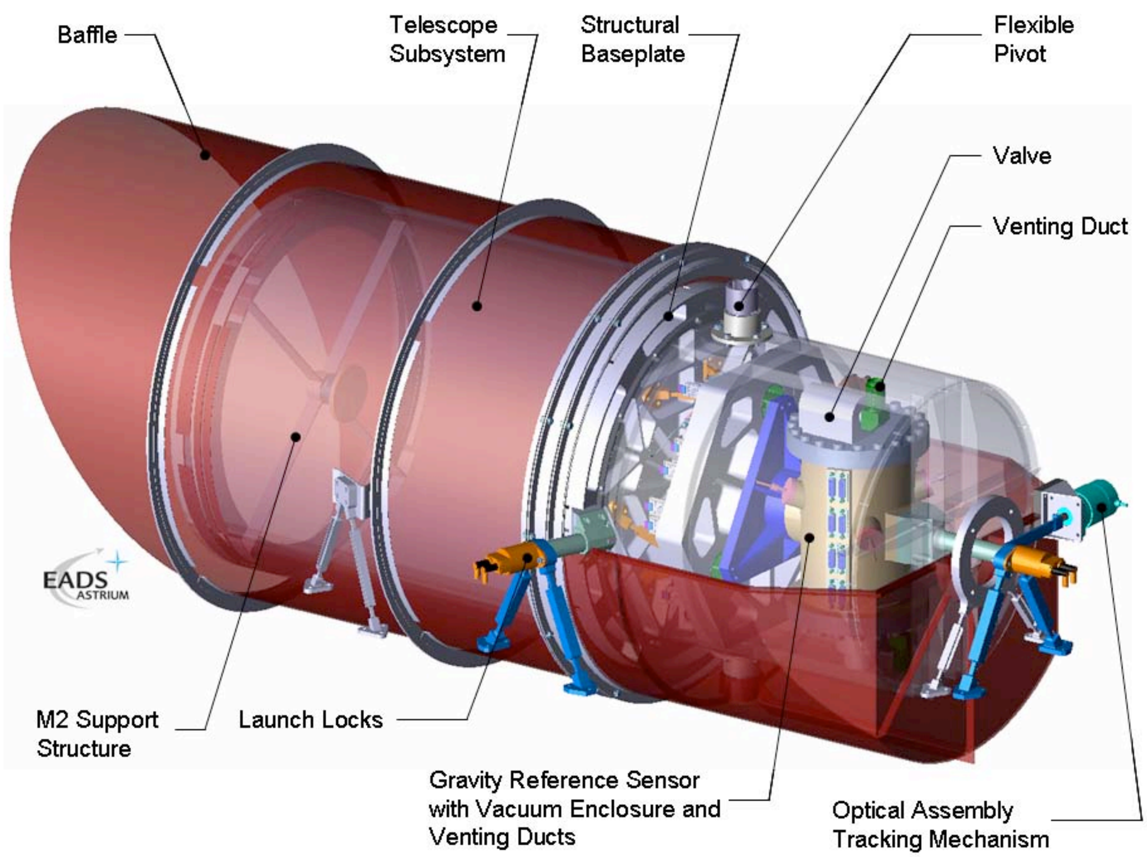


Figure 4.2-1: LOCS Optical Assembly

4.2.2 LOCS Mechanical Interfaces

4.2.2.1 OA Flexible Pivot Bipod Mounts

The LOCS assemblies will be installed on the bottom deck of the Sciencecraft bus using flex pivot bipod mounts as shown in Figure 4.2-1. The flex pivot bipod mounts are required to both support the LOCS assembly and to enable the LOCS assembly to sweep the outbound laser during the laser acquisition stage of the Sciencecraft commissioning phase. Two bipod mounts will be located on either side of the optical bench to provide pitch adjustment. A third bipod mount will be located at the GRS end of the optical assembly to provide yaw motion. More details about the LOCS flexible pivot bipod mounts can be found in the LISA Payload Specification and the LISA Payload Interface Control Document.

4.2.2.2 OA Launch Locks

Launch locks will carry the dynamic load of each OA during launch. As with the flex pivot mounts, two launch locks will be located on either side of the optical bench and a third launch lock will be located at the GRS end of the OA. The use of launch locks to support the telescope secondary mirror spacer is under consideration, however this will be avoided if possible due to reliability concerns. The launch lock configuration is shown in Figure 4.2-2.

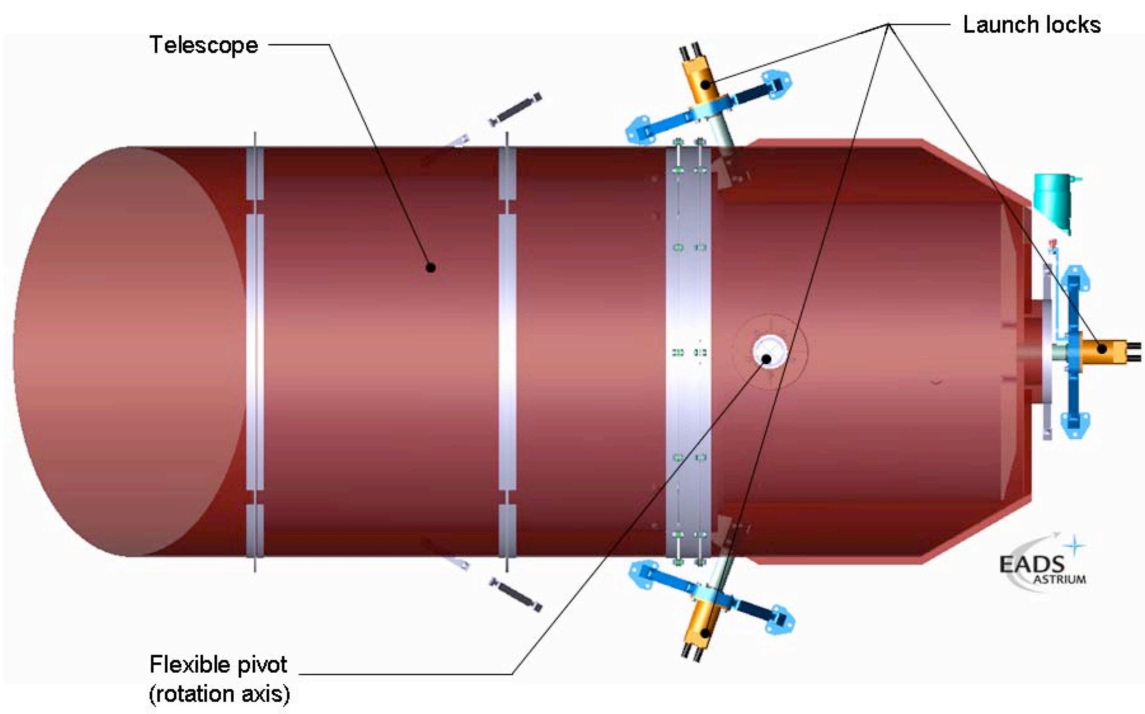


Figure 4.2-2: Launch Lock Configuration

4.2.2.3 Electronics Box Mounting Interfaces

The LOCS/LIMAS electronics boxes will be bolted to the bottom deck of the Sciencecraft bus. This interface will be designed to maximize heat conduction from the electronics boxes through the bottom deck so that it can be radiated out to space. For more information about LOCS/LIMAS electronics box mechanical interfaces refer to the LISA Payload Description Document and the LISA Payload Interface Control Document.

5 Propulsion Module (P/M)

The following section presents a brief overview of the LISA Propulsion Module design and its interfaces to the Sciencecraft. The purpose of this section is to provide a contextual reference for the Sciencecraft design. Detailed performance and interface requirement information for the P/M can be found in the LISA Propulsion Module Description Document and the LISA Propulsion Module Interface Control Document respectively.

5.1 Propulsion Module Overview

Each Sciencecraft will be attached to a P/M from launch up until Sciencecraft commissioning. The P/M provides the following functions:

1. Provide the delta V capability to transfer and orient the S/C from the EELV insertion/separation phase to the required science orbit
2. Mechanically support the Sciencecraft during ground operations
3. Provide the primary load path for the S/C during launch

The P/M design accomplishes the above requirements with a design that includes:

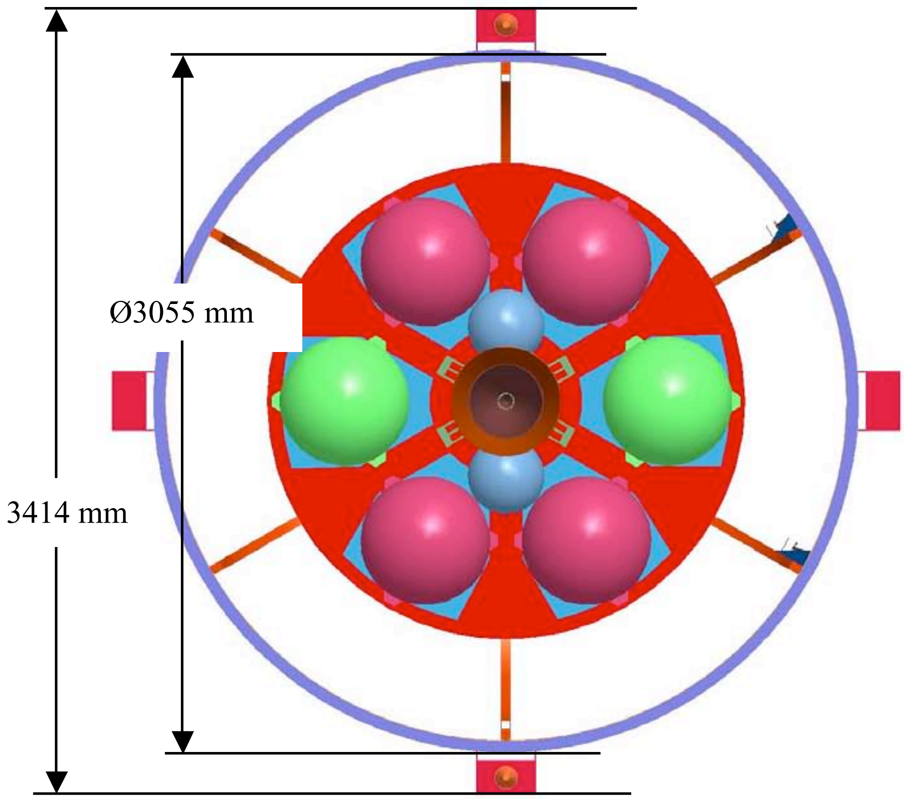
- A structure that supports all of the propulsion subsystem elements, provides a stiff interface with the launch vehicle and supports the SC
- Propulsion subsystem elements (propellant storage, regulation & distribution and thrusters)
- Electrical harnessing linking the propulsion subsystem and thermal hardware internally and to the SC I/F (the SC will provide all power, TC and control functions to the P/M)
- A thermal subsystem (e.g. including but not limited to: MLI, thermal spacers, heaters, paint, thermistors/thermostats etc) to maintain the P/M temperature within acceptable limits
- Coarse Sun Sensors and Star Tracker Camera Head Units to provide position, orientation and rate data to the Sciencecraft ACS.
- Omni Antennas to facilitate communication between the Sciencecraft Comm. System and the ground link during the cruise phase

5.1.1 P/M Propulsion System Design

The P/M Propulsion System is responsible for transferring the S/C from LEOP to science orbit within the 18 month cruise phase. Once the S/C is positioned into science orbit, ACS thrusters will orient the Sciencecraft to the 30 degree angle of incidence to the sun. The ACS thrusters will then impart a spin upon the S/C to give the Sciencecraft axial stability during separation. The function of the propulsion system ends once the Sciencecraft is deployed. The propulsion system baseline design is a dual mode system that employs a single bi-prop Liquid Apogee Engine (LAE) main thruster for ΔV thrust and a dual string array of mono-prop ACS thrusters. The P/M Propulsion System is shown in Figure 5.5-1&2. The P/M Propulsion System components will be mounted on the P/M structure, therefore no mechanical interfaces will exist between the P/M Propulsion System and the Sciencecraft. The electrical interfaces between the P/M and the Sciencecraft will be facilitated by a zero insertion force breakaway connector separation system.

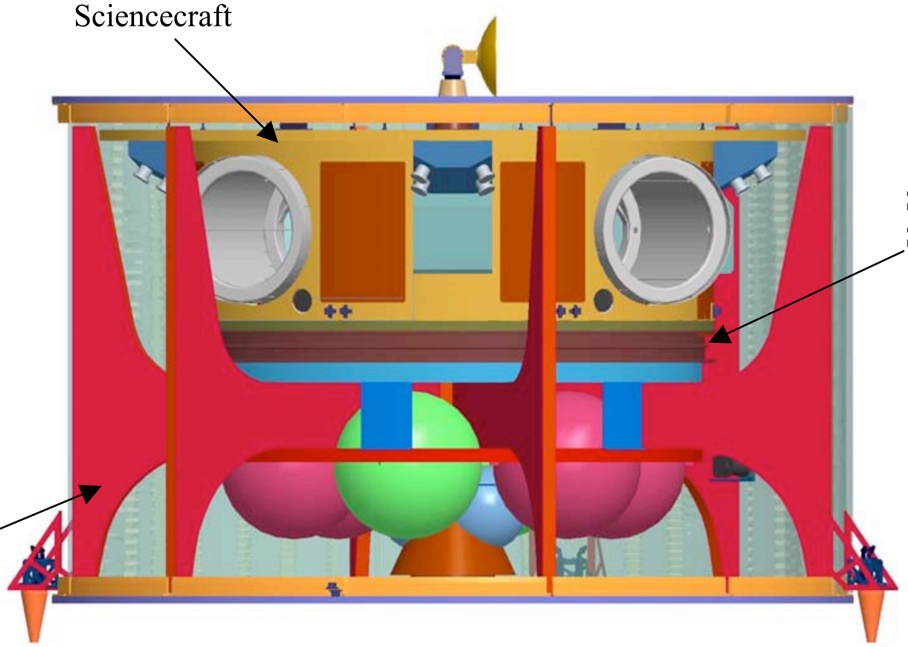
Figure 5.1-1: Propulsion Module (w/o Sciencecraft integrated)

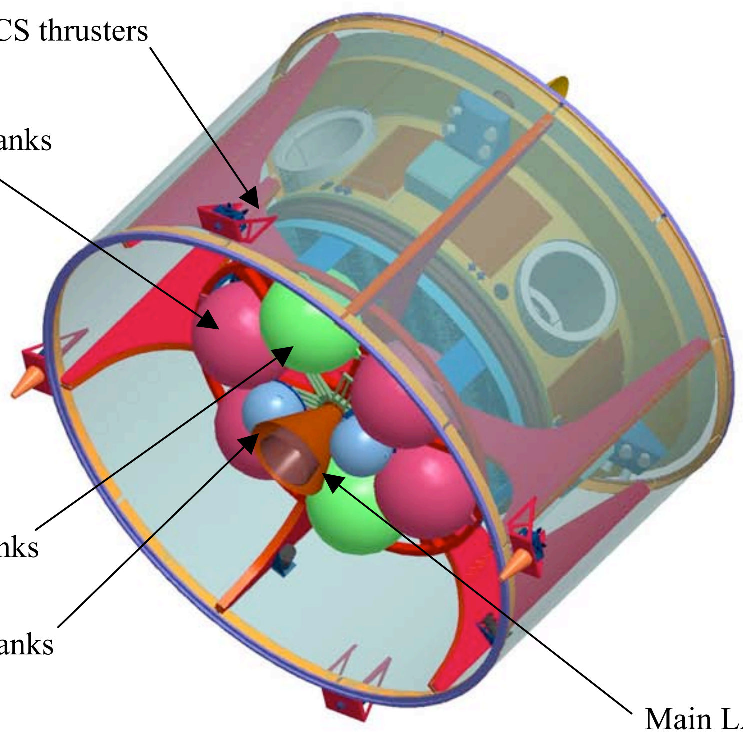
Side View



Top View

Figure 5.2-2: Sciencecraft Integrated into P/M


Sciencecraft
Sciencecraft Separation system
Propulsion Module


ACS thrusters
Hz propellant tanks
NTO oxidizer tanks
He pressurant tanks
Main LAE thruster

5.1.1.1 Sciencecraft Seal Interface

The cleanliness requirements of the payload will require that the Sciencecraft be fully encapsulated by the P/M CST and therefore protected from the launch environment. A gap that is approximately 10cm in size between the Sciencecraft SAD and the P/M CST constitutes an annulus that exposes the Sciencecraft structure and payload of the Sciencecraft to the wider environment of the payload fairing. This exposure applies to the top most Sciencecraft as the bottom two Sciencecraft modules are completely sealed from the fairing environment by virtue of the mated P/M sitting above them. A circular seal placed between the bottom outer edge of the Sciencecraft SAD and a flange on the PM CST interior will be used to provide an enclosed and clean environment for the Sciencecraft once final integration with the PM has taken place.

6 Assembly, Integration, and Testing (AIT)

6.1 Overview

The AIT activities ensure that the payload and spacecraft meet the LISA technical requirements after completion of assembly, integration and testing. The responsibilities and flow for LISA AIT are as follows. ESA develops the three LOCS, verifies them individually and delivers them to JPL. JPL develops the three LIMAS, and integrates them to the three LOCS. After verification that the combined payload functions as a three-arm interferometer JPL delivers the three assemblies to GSFC. GSFC integrates each LOCS/LIMAS assembly onto a spacecraft Bus to form the Sciencecraft. JPL again verifies payload performance prior to and after payload integration with the bus. GSFC will also integrate the PM with each of the three Sciencecraft to form the three spacecraft. After verification that each spacecraft fulfills the performance requirements they are shipped to KSC for launch. A diagram of the overall AIT flow is shown in Figure 6.1-1.

6.1.1 Test Philosophy and Principles

The LISA test philosophy will be that of a Proto-flight (qualification levels/acceptance durations) program, and to test and verify requirements at the lowest level of assembly possible to mitigate and resolve issues as soon as possible. The first LISA spacecraft will undergo more rigorous Proto-Flight qualification testing and performance testing to verify and validate the design. Qualification units will undergo Proto-flight testing and serve as spares as required. As the build progresses to the system level, LISA will test as closely as possible to flight conditions. Subsequent builds (LISA-2 & 3) are tested to acceptance levels. Two important requirements that will be strictly adhered to are the following-

All flight heritage hardware shall be fully qualified and verified for its new application.

All flight electronics hardware shall have a minimum of 1000 hrs accumulated operating/power on operations.

Anomalies encountered during I&T will be documented and investigated to reduce mission risk –

Stop, document, and understand

Purpose of testing is to find problems, not check off a box

Testing will be thorough and complete prior to shipping to KSC for launch vehicle integration.

6.2 PAYLOAD AIT

The LISA payload AIT has two main objectives. First it will bring together the LIMAS subsystem delivered by JPL and the LOCS subsystem delivered by ESA into one payload. Currently, LISA payload integration is done at JPL using the interferometry laboratory. This facility, designed expressly for high-precision interferometry systems, provides for subsystem and system testing. However ESA has recently proposed (Oct 07) an alternate flow to perform LOCS and LIMAS testing and integration at GFSC with a Real-Time Test Bed (RTB).

The interferometric testing is done with the use of a separate test platform that removes seismic disturbances and receives and transmits light to the payload under test. The test platform also includes an optical attenuator and appropriate software to simulate the dispersion and time-delay of the light traversing the 5 million km arm-length. This tests the full interferometric measurement capability of the payload.

6.3 BUS AIT

Bus AIT is performed at GSFC. During this phase the various subsystems such as C&DH, Power, Thermal, Communications, ACS, Mechanisms and Micro-Newton (N) Thrusters are integrated with the core bus. Functional testing is performed to confirm successful integration. The solar arrays are not integrated with the core bus at this time to allow easy access to the LOCS/LIMAS payload, which is next added to the core bus.

6.4 AIT

The addition of the LOCS/LIMAS payload to the core Bus at GSFC defines the Sciencecraft. Functional tests are performed to confirm that that the three Sciencecraft are working individually including interfaces between the payload and other spacecraft subsystems. After integration of the solar arrays to the core bus environmental (thermal vacuum/vibration) tests are performed to check out this configuration that is used for mission operations.

6.4.1 CONSTELLATION (SYSTEM) TESTING

Constellation or System End-to-End testing is to be performed to verify that the three spacecraft are working as a three-arm interferometer. A "range simulator" that provides optical attenuation and emulation of the Doppler shift associated with changes in the on-orbit geometry is used in the testing. Mission operations scenarios and end-to-end communication tests are conducted at this time. Fault detection and correction capabilities are also exercised.

6.5 SPACECRAFT AIT

The addition of the Propulsion Module to the Sciencecraft defines a complete spacecraft. After completion of a new round of functional tests the spacecraft,which is now in its cruise configuration, is returned to the thermal vacuum chamber for additional testing. Other environmental tests for EMI/EMC will be conducted.

6.6 STACK AIT

After the three spacecraft have completed spacecraft AIT they are stacked together for additional functional and environmental testing. Vibration and acoustics testing is performed as well as separation shock tests between spacecraft and between spacecraft and the launch vehicle adapter. The spacecraft then undergo any final rework and preparation for shipment to KSC for launch. Final faring installation will occur in a payload processing facility. Current design fits in a Atlas V 4 meter faring. See Figure 6.1. Fairing model represents recommended maximum static payload envelope

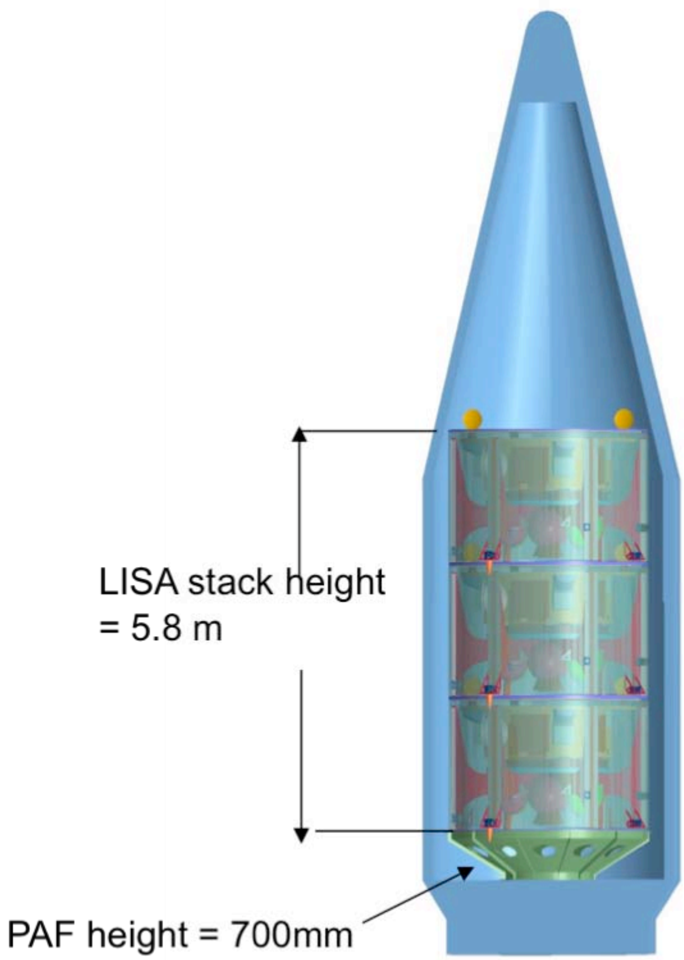


Figure 6.1-1: Atlas V Long 4m faring (XEPF) with LISA SC Stack and PAF

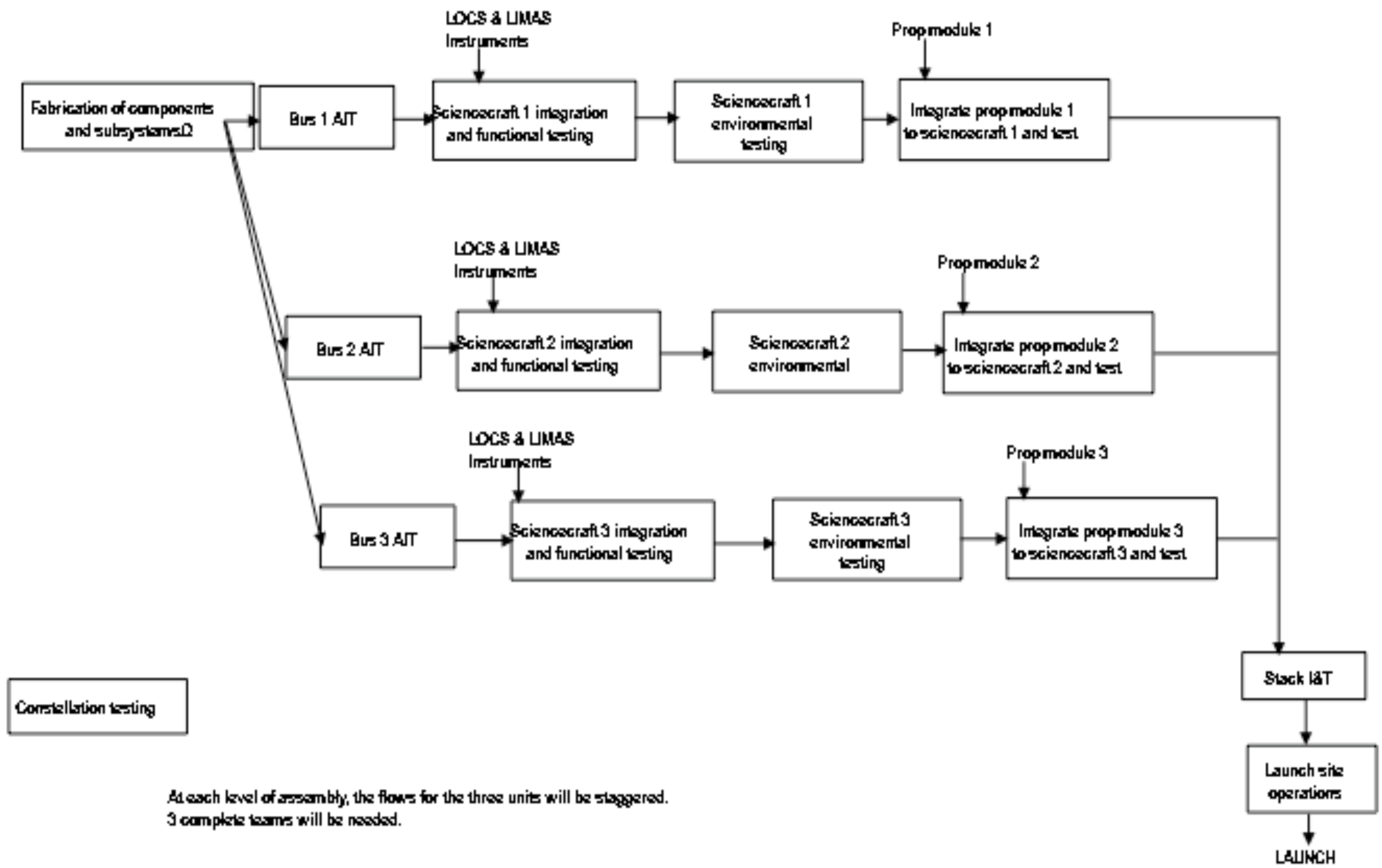


Figure 6.1-2: Top –level AIT

6.7 GROUND SUPPORT EQUIPMENT

Preliminary Ground Support Equipment (GSE) for AIT and other functions has been discussed informally to this point. An initial list will be generated later during Phase A.

NOTE: FINAL DETERMINATION OF REQUIRED GROUND SUPPORT EQUIPMENT (GSE) WILL BE WORKED OUT IN PHASE B.

7 Definitions

The following definitions apply throughout this document:

LISA Scientific Complement/Payload: it includes the LISA Optomechanical Core Systems (LOCS), the LISA Instrument Metrology and Avionics System (LIMAS), the associated control software, micro-thrusters (TBR)

Sciencecraft : one spacecraft Bus with its LISA Scientific Complement

Constellation: the three LISA operating together on-orbit

Bus: the portion of the Sciencecraft that supports (functionally/mechanically) the payload

Propulsion Module: component that provides support for the Sciencecraft during launch and provides the delta-v during the cruise phase of the mission

Spacecraft: the combination of the Sciencecraft and the Propulsion Module also known as the "cruise configuration".

Stack: the integrated launch "stack" of 3 S

8 ACRONYMS

ACS	Attitude Control System
ADC	Analog to Digital Converter
AIT	Assembly, Integration, and Test
AIVT	Assembly, Integration, Verification and Test
AST	Analog Star Trackers
AU	Astronomical Unit
CBOD	Clamp Band
CCD	Charge-Coupled Device
C&DH	Command & Data Handling
cFE	Core Flight Executive
CFLR	Centaur Forward Load Reactor
CFRP	Carbon Fiber Reinforced Plastic
CFS	Core Flight System
CHU	Camera Head Unit
CMS	Charge Management System
CSS	Coars Sun Sensor
CTE	Coefficient of Thermal Expansion
CMNT	Colloidal Micro-Newton Thrusters
CPB	CPU Power Block
DAC	Digital to Analog Converter
DFACS	Drag Free Attitude Control System
DFC	Drag Free Control
DoD	Depth of Discharge
DPLL	Digital Phase Locked Loop
DRS	Disturbance Reduction System
DSN	Deep Space Network
DTM	Deterministic Transfer Maneuver
DSS	Digital Sun Sensors
EADS	European Aeronautic Defence and Space Company
EM	Engineering Model
EOL	End of Life
ESA	European Space Agency
FDIR	Failure Detection Isolation and Recovery
FM	Flight Model
FoV	Field-of-View
FTR	Final Technical Report
FPA	Fiber Power Amplifier
GsAs	Gallium Arsenide
GRS	Gravitational Reference Sensor
HGA	High Gain Antenna
KSC	Kennedy Space Center
FEEP	Field Emission Electric Propulsion

FPGA	Field Programmable Gate Array
FWHM	Full Width Half Maximum
FSW	Flight Software
H/W	Hardware
ICD	Interface Control Document
IMS	Interferometry Measurement System
IWS	Inertial Wavefront
JPL	Jet Propulsion Laboratory
LEOP	Launch and Early Operations
LGA	Low Gain Antenna
LIMAS	LISA Instrument Metrology and Avionics System
LiIon	Lithium Ion
LNP	Low Noise Amplifier
LO	Local Oscillator
LOC	Lines of Code
LOCS	LISA Optomechanical Core Systems
LOS	Line of Sight
LPF	LISA Pathfinder
LTP	LISA Technology Package
MDL	Mission Design Lab (GSFC)
MOC	Mission Operations Center
MOFA	Master Oscillator Fiber Amplifier
MOPA	Master Oscillator Power Amplifier
MoU	Memorandum of Understanding
MRD	Mission Requirements Document
Nd:YAG	Neodymium doped Yttrium Aluminum Garnet
NEA	Noise Equivalent Angle
nm	Nano-metre (10^{-9}m)
NPRO	Non-Planar Ring Oscillator
OA	Optical Assemby
OB	Optical Bench
OATM	Optical Assembly Tracking Mechanism
OSR	Optical Surface Reflectors
PAA	Point Ahead Actuator
PAF	Payload Adapter Fitting
PL	Payload
PLF	Payload Fairing
pm	pico-metre (10^{-12}m)
PM	Proof Mass
P/M	Propulsion Module
PV	Photo Voltaic
QPD	Quad Photo Diode
SA	Solar Array
S/C	Spacecraft
ScRD	LISA Science Requirement Document
SDPS	Science Data Processing Segment

SIB	Spacecraft Interface Block
SNR	Signal to Noise Ratio
SRS	Shock Response Spectrum
SSPA	Soild State Power Amplifier
TBD	To be Determined
TBR	To be Resolved
TBS	To be Supplied
TCM	Trajectory Correction Maneuver
TDI	Time Delayed Interferometry
TM/TC	Telemetry/Telecommand
TWTA	Traveling Wave Tube Amplifier
RMS	Root Mean Square
UPB	Unswitched Power Block
USO	Ultra Stable Oscillator
UV	Ultra Violet
wrt	…with respect to
Yb:YAG	Ytterbium doped Yttrium Aluminium Garnet

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